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Architecture analysis of wireless power transmission for lunar outposts

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MONTEREY, CALIFORNIA

THESIS

ARCHITECTURE ANALYSIS OF WIRELESS POWER TRANSMISSION FOR LUNAR OUTPOSTS

by

William T. Reynolds

September 2015

Thesis Advisor:	Charles Racoosin
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To continue scientific research on the moon, largely abandoned since the Apollo era, humanity must establish a permanent outpost. This research has narrowed the lunar base sites to the polar regions as these sites offer the highest scientific value. The overarching problem is how to supply continuous power to lunar bases located at the poles. This study focuses on the feasibility and architectural analysis of wireless power transfer to lunar polar outposts. Two wireless power transfer methods, microwave and laser, were integrated into satellite constellations and the overall system architecture. The two architectures were modeled, analyzed, and evaluated to determine which method is more feasible. The results showed that while both the use of microwave and laser transmission were feasible, the microwave approach produced large transmitter and receiver antenna sizes driving unreasonable cost. The laser transmission approach showed less end-to-end efficiency and therefore higher per satellite cost but resulted in a lower total system cost and was the more feasible architecture.

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ARCHITECTURE ANALYSIS OF WIRELESS POWER TRANSMISSION FOR LUNAR OUTPOSTS

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ABSTRACT

To continue scientific research on the moon, largely abandoned since the Apollo era, humanity must establish a permanent outpost. This research has narrowed the lunar base sites to the polar regions as these sites offer the highest scientific value. The overarching problem is how to supply continuous power to lunar bases located at the poles. This study focuses on the feasibility and architectural analysis of wireless power transfer to lunar polar outposts. Two wireless power transfer methods, microwave and laser, were integrated into satellite constellations and the overall system architecture. The two architectures were modeled, analyzed, and evaluated to determine which method is more feasible. The results showed that while both the use of microwave and laser transmission were feasible, the microwave approach produced large transmitter and receiver antenna sizes driving unreasonable cost. The laser transmission approach showed less end-to-end efficiency and therefore higher per satellite cost but resulted in a lower total system cost and was the more feasible architecture.

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LIST OF ACRONYMS AND ABBREVIATIONS

AEHF	advanced extremely high frequency
a.u.	astronomical unit
DARPA	Defense Advanced Research Projects Agency
ESA	European Space Agency
EVA	extravehicular activity
ft	feet
GHz	gigahertz
GW	gigawatt
HCT	hall current thrusters
ISC	Integrated Symmetrical Concentrator
ISS	International Space Station
JAXA	Japan Aerospace Exploration Agency
JSC	Johnson Space Center
kg/m	kilogram per meter
kg/m2	kilograms per meter squared
kHz	kilohertz
km	kilometers
kW	kilowatts
kW-hrs	kilowatt-hours
kW/kg	kilowatts per kilogram
LPI	Lunar and Planetary Institute
MHz	megahertz
MW	megawatt
mT	metric Ton(s)
NASA	National Aeronautics Space Administration
nm	nanometers
NRC	National Research Council
PGT	platform generic technologies
PMAD	power management and distribution
RF	radio frequency

SERT	SSP Exploratory Research and Technology
SLA	stretched lens array
SLS	space launch system
SPA Basin	South Pole-Aitkin Basin
SPG	solar power generation
SPS	solar power satellite
SPS-ALPHA	Solar Power Satellite via Arbitrarily Large Phased Array
SSP	space solar power
STK	systems tool kit
TLI	trans-lunar injection
TMS	thermal management system
VMJ	vertical multi-junction
W/cm^2	watts per centimeter squared
W-hr/kg	watt-hour per kilogram
W/kg	watts per kilogram
WPT	wireless power transmitter

EXECUTIVE SUMMARY

In order for people to more fully explore the Moon, a continuous supply of electrical power would be required. The primary research was to determine if it is feasible to provide power to a lunar polar outpost using a satellite constellation in lunar orbit. To answer this question, it was necessary to conduct a literature review on several aspects of the problem including the location of a lunar outpost, the types of lunar orbits, the history and state of the technology for microwave and laser wireless power transmission, and the history and development of solar power satellite concepts. The location of the outpost was derived from the work done by NASA on the Scientific Context for Lunar Exploration and a landing site survey (Kring 2012). The two options for the outpost location with the highest scientific value were at the South Pole and the South Pole-Aitken Basin. The lunar orbit type applicable to solar power satellites included frozen elliptical orbits. The frozen elliptical orbit is a class of stable, highly inclined elliptical orbits for polar regions with a ten-year duration that require little station-keeping, first examined by Todd A. Ely (Ely 2005). Ely's research into these orbits produced a three-satellite constellation that provided continuous coverage of the lunar South Pole (Ely 2006).

Microwave wireless power transmission is the current default choice for wireless power transfer and a significant amount of scientific research, development, and testing has been conducted to advance the use of this frequency range. In the 1970s, NASA in conjunction with the Department of Energy conducted tests using the extremely high frequency (EHF) portion of the electromagnetic spectrum for wireless power transfer which would become the standard approach for microwave wireless power transfer (Brown 1984). The standard approach included the use of a parabolic transmitter pointed at a receiving array. Laser wireless power transmission enjoys a long history of both theoretical and material origin, as Max Plank formulated the underlying idea that light was just another form of electromagnetic radiation and received the Nobel Prize for it 1918 (Hecht 1992). Building on the theoretical work done by several scientists, including Albert Einstein, Theodore Maiman, in May of 1960 created the first operating laser at

Hughes Research Laboratory in California (Maiman 1960). The physics of laser wireless power transmission provided the calculations, and since the laser will propagate through a vacuum it can be treated as a Gaussian beam. A Gaussian beam is used in optics when a beam of electromagnetic radiation has an electric field and intensity profiles in the transverse plane that can be approximated by Gaussian functions (Svelto 1976).

The first person to develop the idea of using satellites for power generation was Dr. Peter Glaser, in 1968, with his concept of a Solar Power Satellite (SPS). His concept was for a large platform placed in geosynchronous orbit, which collected the continuous solar energy of the Sun and utilized wireless power transmission to beam solar energy via microwaves to Earth's receiving stations (Glaser 1968). Further research identified five solar power satellite concepts including the 1979 Reference Design, the SunTower, the Integrated Symmetrical Concentrator, the First International Assessment of Space Solar Power, and the SPS-ALPHA concepts. These concepts provided useful figures of merit and analysis methodology used in this thesis.

The results of these previous efforts were then used to select an appropriate lunar outpost location and develop and analyze lunar orbits using the Systems Tool Kit (STK) software suite. After identifying the lunar outpost site, power requirement, and constellation parameters, a system cost, mass, and power thread calculation was conducted on two wireless power transmissions options using microwave or laser power beaming. Finally, a sensitivity analysis was conducted on key figures of merit in the architecture to highlight areas that need further technological development to increase end-to-end efficiency and/or decrease potential system cost. The end goal of this thesis was to set up a cost comparison to determine which wireless power transmission option was more feasible.

The lunar outpost location was the first parameter selected. The location of the outpost was derived from the work done by NASA on the Scientific Context for Lunar Exploration and a landing site survey. The two options for the outpost location were at the South Pole and the South Pole-Aitken Basin. From the landing site survey the SPA Basin offered the higher scientific value and was chosen as the outpost location. The next parameter selected was the type of lunar orbit that provided coverage to these outpost

locations. The choice of orbit types were Halo orbits at the L1 or L2 libration point, low lunar orbits, and frozen elliptical orbits. The frozen elliptical orbit was selected for analysis as it offered the best coverage for the SPA Basin and can be seen in Figure 1. The resulting satellite constellation had a 20˚ minimum elevation angle constraint to the lunar outpost, with four total satellites with two-satellites in two-planes each. The planes were outset by 180˚ with a worst case distance from satellite to lunar outpost of 9881 kilometers.

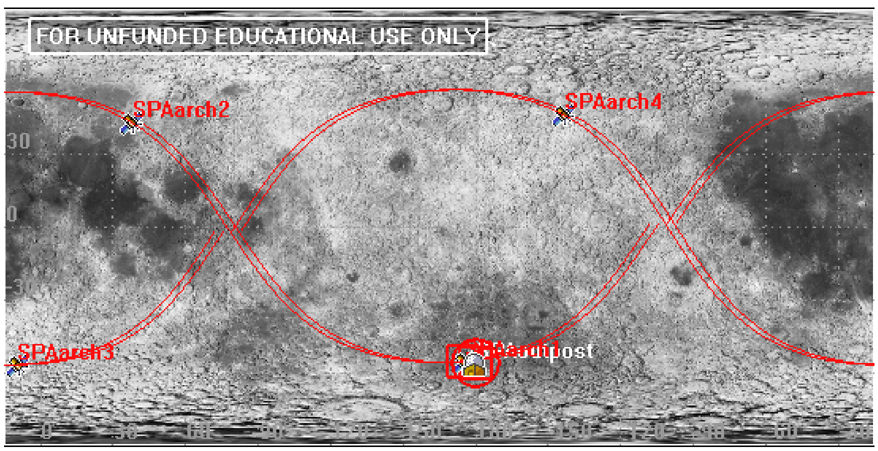

Figure 1. Ground Track for 4 Satellite Constellation for SPA Coverage

The lunar power requirement was derived from the International Space Station. The ISS solar wing arrays produce between 84–120 kilowatts (kW) with a standard crew size of three and maximum long-duration crew size of six. A 100-kilowatt power requirement was used for analysis of the SPS concepts.

With the outpost location, orbit type, and constellation parameters identified, a modified version of the First International Assessment of Space Solar Power was used. This methodology required the development of a functional architecture and identifying the interrelationship of element in the functional architecture. The key to this type of methodology is finding relevant figures of merit for architecture elements. The literature review on wireless power transfer and other solar power satellites concepts provided these figures of merit. The functional architecture and interrelationships can be seen in Figure 2 and Figure 3.

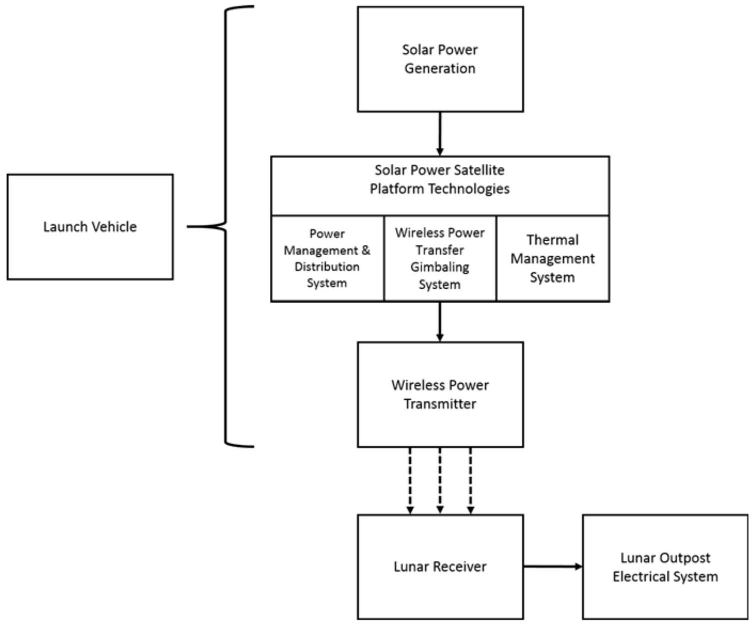

Figure 2. SPS Functional Architecture

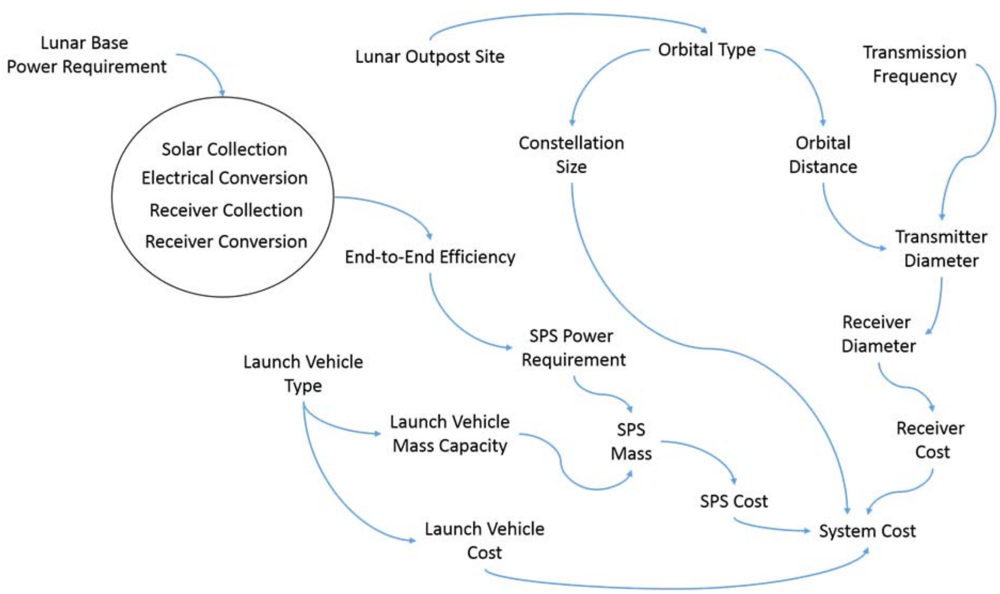

Figure 3. Relationship of Functional Elements

From the functional architecture and understanding the relationship between the elements a mass, cost, and power thread calculation was done for microwave and laser wireless power transmission using the SPA Basin outpost location, frozen elliptical orbit, and lunar outpost power requirement parameters. The results of these calculations are shown in Table 1. The result of these calculations using total system cost as the

determining criteria was a microwave SPS total system cost of $3,425 million and a laser SPS total system cost of $3,078 million. While both the microwave and laser SPS concepts were feasible, the laser SPS concept achieved a lower cost and is the more feasible system.

Table 1. Microwave and Laser SPS Comparison

	MICROWAVE	LASER	UNITS
FREQUENCY	94		GHz
WAVELENGTH		805	Nanometers (nm)
LUNAR OUTPOST POWER REQUIREMENT	100	100	kW
SATELLITE RANGE	9881	9881	km
END-TO-END EFFICIENCY	11 %	8.3 %	%
SATELLITE POWER REQUIREMENT	912	1208	kW
SATELLITE POWER PER UNIT MASS	0.2	0.2	kW/kg
SATELLITE MASS	4.6	6.0	Metric Tons (mT)
SATELLITE COST PER UNIT MASS	100,000	100,000	$/kg
SATELLITE COST	456	604	$M
TRANSMITTER DIAMETER	277	5.45	Meters
RECEIVER DIAMETER	278	21	Meters
RECEIVER COST	970	30	$M
RECEIVER LUNAR COST PER UNIT MASS	100,000	100,000	$/kg
TOTAL LAUNCH VEHICLE COST	632	632	$M
LAUNCH VEHICLE MASS TO TLI	13.2	13.2	Metric Tons (mT)
CONSTELLATION SIZE	4	4	Satellites
TOTAL SYSTEM COST	3,425	3,078	$M

After completing the mass, cost, and power thread calculations a sensitivity analysis was conducted to understand how varying parameters affected the total system cost of the microwave and laser SPS concepts. Since the microwave lunar receiver was the driving cost factor over the laser system the cost to land mass on the Moon was allowed to vary from $100,000 per kilogram to $50,000 per kilogram. The results of this sensitivity analysis showed that the per-kilogram cost to land on the Moon needed to be below $62,500 to be more economically feasible with the laser SPS concept. The final conclusion after the sensitivity analysis was that the Laser SPS concept was the more feasible solution and could provide continuous power to a lunar outpost.

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I. INTRODUCTION

A. CONTEXT

Eugene Cernan was the last man to step foot on the Moon in completion of the Apollo mission in December 1972 (Cernan 1999). Since that time, no other human has made contact with the Moon's surface. However, this status quo is about to change, given the recent re-invigoration of National Aeronautics and Space Administration (NASA) as an exploration agency, and advances in commercial launch vehicles. While many in the space exploration business wish to go beyond the Moon, to Mars and to the asteroids, the technology for these ventures is not yet adequate for the task. Given this restriction, this thesis assumes the next space exploration mission will be to establish a permanent lunar outpost in the polar regions of the Moon which has the potential to be accomplished.

1. Lunar Exploration

In September 1962, President Kennedy posed the question, "But why, some say, the Moon? Why choose this as our goal? We choose to go to the Moon … not because [it is] easy, but because [it is] hard; because that goal will serve to organize and measure the best of our energies and skills, because that challenge is one that we are willing to accept, one we are unwilling to postpone, and one which we intend to win" (Kennedy 1962). This answer is still as satisfying 53 years later as it was in 1962; however, while the Apollo mission saw success, it never lived up to the overarching dream it had inspired. That dream was to see mankind fully explore and establish a continuous presence on the Moon in perpetuity as an example of mankind's inexorable charge into space. The Moon, given its accessibility, is a natural next step for human settlement on mankind's journey toward Mars, and as a testbed for the technologies and concepts that will be needed for long-term human survival in space. The lunar environment offers several exceptional prospects for continued research of the Earth and of the greater universe. Lunar research opportunities include the study of space weather, Earth's albedo or, given the shielding influence of the Moon, radio astronomy mapping of the universe on the far side. Additional lunar research is the study of the Moon itself, including the origin of Earth's

only satellite, its evolution, composition, and internal structure. In both NASA's Vision of Space Exploration (Newman 2005) and the European Space Agency's (ESA) MoonNEXT mission, the objective was the scientific exploration of the permanently shadowed areas around the lunar poles (Carpenter 2008). These craters offer the best possible conditions for infrared observations due to their continuous darkness and low temperatures. Also, the lunar poles offer new locations for *in situ* crust and mantle research not conducted during the Apollo missions and has the greatest probability of researchers finding ice on the Moon.

One of the many challenges to establishing a permanent lunar outpost is the need to provide continuous sustainable power. A useful reference point to establish the amount of power needed to maintain a lunar outpost can be derived from the space community's experience with the International Space Station (ISS). The ISS solar wing arrays produce between 84–120 kilowatts (kW) with a standard crew size of three and maximum long duration crew size of six (Wells 2009). One method of providing the power needed for a lunar polar outpost is the use of a solar power satellite.

2. Solar Power Satellite

The first person to develop the idea of using satellites for power generation was Dr. Peter Glaser in 1968 with his concept of a Solar Power Satellite (SPS). His concept was for a large platform placed in geosynchronous orbit, which collected the continuous solar energy of the Sun and utilized wireless power transmission to beam solar energy via microwaves to Earth's receiving stations (Glaser 1968). This design has several advantages to terrestrial-based solar power generation methods, as this design avoids the impediments of inclement weather and night time outages during periods when no solar energy can be collected. This design would produce a steady supply of continuous energy, except during small periods when the SPS was in eclipse, and would achieve a higher energy efficiency than solar power systems based on the ground. The fundamental goal of Glaser's design was to provide a supply of clean, renewable, and affordable power in the face of ever increasing demand for energy and the subsequent environmental costs from current fossil fuel power generation (1968).

3. Power Generation for Lunar Exploration

There are several options for generating power for a lunar outpost and lunar exploration. These options include the use of photovoltaics directly on the Moon, the use of nuclear power, and the use of a solar power satellite. This thesis will provide a review of each option and discuss the issues concerning the use of photovoltaics and nuclear. The use of photovoltaics directly on the Moon has two potential issues. The first issue is the available sunlight for a given lunar outpost location, where some locations receive light less than half of the time, and therefore would require large storage batteries (Brinker 1988). The first issue with photovoltaics and large storage batteries can be dealt with using peaks of eternal light, which are mountains that receive continuous light given the slight tilt in the Moon's orbit (Kruijj 2000). The use of peaks of eternal light reduces the energy storage system mass required for surface photovoltaic systems that experience less available sunlight. However, this leads to a second issue: photovoltaics, positioned on peaks of eternal light, would require the use of power cables to transport energy to lunar outposts (Popovic 2008). The other option for generating power requires the use of nuclear fission. This option has two issues concerning safety and mass. While nuclear type payloads have been launched in the past, a nuclear reactor large enough to power a lunar outpost in the hundreds of kilowatts would pose a significant launch risk and exposure of the environment to nuclear contamination from a failed launch or lunar landing (Achenbach 2015). In addition, a nuclear power generating plant for a lunar outpost would create a safety issue for landing on the Moon and to any lunar astronauts who would have to maintain this equipment. These concerns lead to the final option, which is the use of a solar power satellite wirelessly beaming power to a lunar outpost.

B. PURPOSE AND RESEARCH QUESTION

This research will propose, analyze, and compare two different electromagnetic architectures for powering lunar polar outposts using solar power satellites. Based on this research, lunar outposts could exist in the polar regions of the moon allowing a continuous human presence on the moon for scientific research of the Earth, Moon, and

the entire universe. The question this research will attempt to answer: Is it feasible to provide power to a lunar polar outpost using a satellite constellation in lunar orbit?

C. SCOPE, LIMITATIONS, AND METHODOLOGIES

There are two parts to the scope of this research. First, this thesis surveys and provides a summary of the literature for the most current technologies for wireless power transfer. Second, given these technologies, satellite architectures are developed using two electromagnetic options. These two wireless power and satellite architectures are then analyzed for end-to-end performance for providing power to lunar polar outposts. The methodology of this research includes a literature review of wireless power transfer, available lunar orbits, as well as the scientific context for the location of the lunar outpost. The knowledge gained from the literary review is then used to select an appropriate lunar outpost location, develop and analyze lunar orbits using the Systems Tool Kit (STK) software suite, and run a mass, cost, and end-to-end performance analysis using the outpost location and selected orbit. Finally, a sensitivity analysis is conducted on key figures of merit in the architecture to highlight areas that need further technological development to increase end-to-end efficiency and/or decrease potential system cost.

D. CHAPTER SUMMARY

Chapter I serves as an introduction to the thesis. Background information is included in Chapter II to familiarize the reader with the numerous concepts that will be used in the following chapters for analysis. The background information includes the scientific context for the exploration of the Moon, an overview of orbital motion and its lunar application, wireless power beaming technologies and options, information on the history and development of space based solar power, and non-satellite options for lunar power generation. This background information provides the reader the boundaries and theoretical details in which to determine the feasibility of a solar power satellite powering a lunar polar outpost.

Chapter III identifies a theoretical model to conduct the analysis and to determine the feasibility of the system. Using the architectures defined in Chapter III, Chapter IV

explains the location of the lunar outpost and shows coverage to applicable lunar orbits modeled in STK. The STK results and the methodology used to develop these orbits is presented followed by graphical outputs from STK. Lastly, with the lunar outpost location and satellite constellation selected, an analysis of the mass, cost, and power thread using key figures of merit for solar power satellites for each model is presented. In addition to the model for each solar power satellite, sensitivity analysis is conducted to show the boundaries for key figures of merit to produce a feasible system solution.

A general summary of the background information and analysis conducted is provided in the final chapter. The thesis question is addressed and conclusions about the feasibility of the proposed system solutions are drawn using the findings from Chapter IV. Chapter V concludes with suggested areas of future work that could improve the feasibility and/or analysis fidelity of proposed systems.

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II. BACKGROUND

The application of a space-based solar power system to generate and transmit power to a lunar outpost located in its polar regions requires a thorough explanation of the physical parameters and limitations of the technologies that influence the system architecture and therefore system feasibility. The location of the outpost site and the relative satellite motion of stable lunar orbits are critical system architecture factors. In addition, the amount of power required by the lunar outpost influences the spacecraft mass and how the satellite generates and transmits energy are all vital discussion points. Finally, with the identification of the lunar outpost site, outpost power requirement, stable lunar orbit, and solar power satellite design completed, this section will review the technology for the lunar outpost receiver. This chapter will identify the key parameters of the system architecture that will provide context for a solution presented in later chapters of this research paper.

A. LUNAR ORBITAL MOTION

Modern astronomy has a long history of development starting with a change in paradigm from the Greek Earth-centered universe to the Sun-centered solar system developed by Copernicus and improved by Johannes Kepler in the early 1600s. Kepler, using the calculations of Nicolaus Copernicus and the observations of Tycho Blahe, explained the elliptical motion of the planets and therefore other bodies in orbit and recorded this updated paradigm in his laws of planetary motion (Sellers 2000). The first law changed the orbit of each planet to an ellipse and moved the Sun to one of the focal points. The second law created an imaginary line connecting a planet to the Sun that sweeps out equal areas in equal times. The third law stated that the period of the planet squared is proportional to the mean distance from the Sun cubed (Sellers 2000).

Kepler’s paradigm was very successful in describing the celestial mechanics of the planets, but it provided no explanation for “why” the planets moved in this manner (Sellers 2000). It was Isaac Newton who incorporated the work of Galileo Galilei and his study of terrestrial dynamics and Kepler’s insight to form the laws of motion (Sellers

2000). Newton's three Laws of Motion and his Universal Law of Gravitation provided both a description and a mathematical explanation for the motion of the planets and therefore the movement of objects in the solar system. Newton's first law of motion is the law of inertia, which states that a body remains at rest or moves in a straight line unless acted upon by an outside force. The second and third law of motion, is the law of action and reaction. The second law of motion was Newton's true insight which describes how a force is equal to the product of a body's mass times its acceleration (Capderou 2006). Newton's laws provided the underlying knowledge for the study of the motion of celestial objects which became known as astrodynamics. Astrodynamics allows the accurate portrayal of orbiting bodies. Those orbits can be described as elliptical trajectories that are periodic in nature with a central planetary object as the reference frame (Montenbruck 2012).

Lunar orbits, called selenocentric orbits, refer to the orbit of a body around the Moon with respect to its center. There are two types of selenocentric orbits called low lunar orbit and frozen orbit. Low lunar orbits are defined as orbits below 100 kilometers in altitude. The application of this orbit is particularly valuable in the exploration of the Moon; however, uneven gravitational perturbations caused by the Moon make many unstable (Meyer 1994). These gravitational perturbations are caused by the mass concentrations, called mascons, below the lunar surface and result in changes to selenocentric orbits within several days (Meyer 1994).

In 2001, it was discovered that the effect of these mascons on lunar satellites could be used to create frozen orbits (Bell 2006). Frozen orbits are defined as a type of orbit where due to the shape of the central body and wisely selecting the initial orbital parameters, the satellite's altitude is always at the same point in each orbit (Bell 2006). The gravitational perturbations cancel out the changes to the satellite's inclination, apogee, and eccentricity. The results are an orbital type that is stable over the long term with minimal station-keeping maneuvers (Folta 2006). Frozen orbits can be used for low lunar orbits and have also been modeled at higher altitudes against real operational data. An example of this comparison against real operational data was the Clementine spacecraft launched in 1994, the modeling results matched the operational data in the

predicted pattern (Folta 2006). The Clementine spacecraft had an apoapsis of 3000 kilometers and a periapsis of 400 kilometers. Low lunar frozen orbits have been found at several orbital inclinations to include 27˚, 50˚, 76˚, and 86˚ degrees (Bell 2006). Higher frozen orbits have been found at orbital inclinations of 40˚, 45˚, and 56˚ degrees with a minimal altitude of 225 kilometers (Ely 2005). Frozen orbits unique characteristics allow a greater utility than low lunar orbits and are useful for science, communication, and navigation missions.

Of particular interest to this thesis topic is a class of stable, highly inclined elliptical orbits for polar regions with a ten-year duration that require little station-keeping, first explored by Todd A. Ely (2005). This orbit type is a complex three-body dynamic problem in which satellites in orbit experience significant perturbations when at altitudes greater than 500 km to 20,000 km from the Earth's gravitation with secondary perturbations from the Sun, the non-spherical gravity field of the Moon, and solar radiation pressure (Ely 2005). After modeling and solving for the effects of these perturbations, a unique orbit was found for the elliptical inclined orbit where the perturbations created a bounded quasi-periodic behavior without transferring the satellites in these orbits onto hyperbolic or chaotic trajectories (Ely 2006). Figure 1 illustrates a five-year propagation with elliptical inclined orbits in red, circular inclined orbits in blue, and circular polar orbits in green. As seen in Figure 1, the elliptical inclined orbit in red, the propagation of the satellite in this orbit is almost periodic while remaining stable.

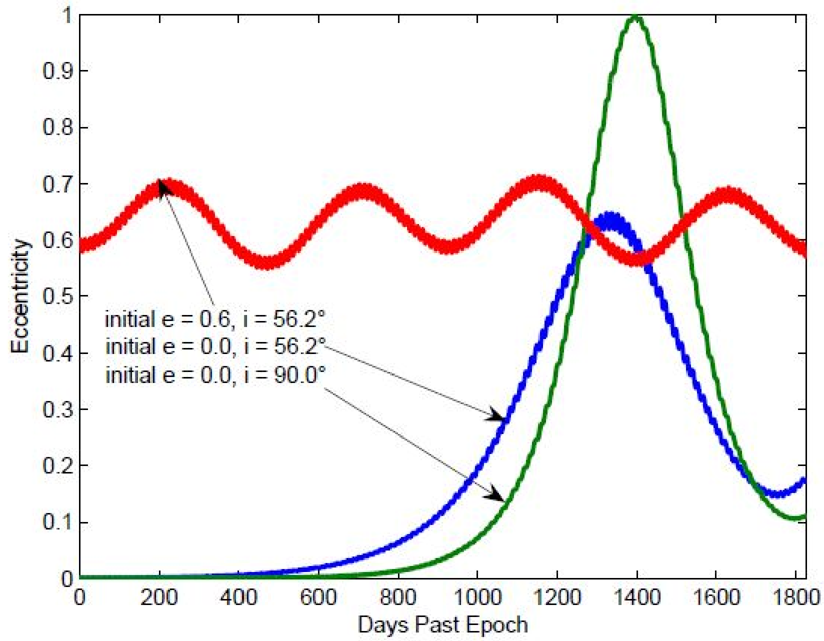


Figure 1. Five-Year Propagations of Elliptical Inclined Orbit, a Circular Inclined Orbit, and a Circular Polar Orbit (from Ely 2006)

From the elliptical inclined orbit, the following orbital elements and coverage were found with a semi-major axis of 6541.4 kilometers (km), an eccentricity of 0.6, an inclination of 56.2˚, ascending node of 0˚, an argument of periapsis of 90˚, and a mean anomaly spacing of 120˚ (Ely 2006).

$$\begin{aligned}\{a_1, e_1, i_1^{op}, \Omega_1^{op}, \omega_1^{op}, M_1\} &= \{6541.4km, 0.6, 56.2°, 0°, 90°, 0°\} \\ \{a_2, e_2, i_2^{op}, \Omega_2^{op}, \omega_2^{op}, M_2\} &= \{6541.4km, 0.6, 56.2°, 0°, 90°, 120°\} \\ \{a_3, e_3, i_3^{op}, \Omega_3^{op}, \omega_3^{op}, M_3\} &= \{6541.4km, 0.6, 56.2°, 0°, 90°, 240°\}\end{aligned} \quad (1)$$

The coverage statistics for this constellation are shown in Table 1 and when using a three-satellite constellation for lunar South Pole coverage results in continuous coverage often with two-fold coverage of the South Pole.

Table 1. Coverage Statistics of a South Polar Station by each Spacecraft and Constellations (from Ely 2005)

Coverage Statistics of a South Polar Station by each Spacecraft and Constellation

	Satellite 1	Satellite 2	Satellite 3	Constellation	Constellation
				1-Fold Coverage	2-Fold Coverage
Mean Pass (hrs)	10.572	10.582	10.579	Total Span	Total Span
Mean Gap (hrs)	3.513	3.507	3.509	0	0
% Coverage	73.35	73.399	73.375	100	100

A type of orbit that is not selenocentric but has been used in space solar-power architectures is the halo orbit. Halo orbits utilize Lagrange points, which were discovered in 1772 by Joseph-Louis Lagrange and published in his report, "Essay on the Three-Body Problem." Lagrange points exist in any three-body dynamic system where a relatively small object is stable between two large masses. The two large masses create zones of equal gravitational potential energy where small objects can "park"; essentially, the gravitational pull of two large masses cancels out.

Figure 2 shows the Lagrange points for the Earth-Moon system where the L1 and L2 are on the near-side and far-side of the Moon at a distance of approximately 57,000 kilometers (Kare 2004).

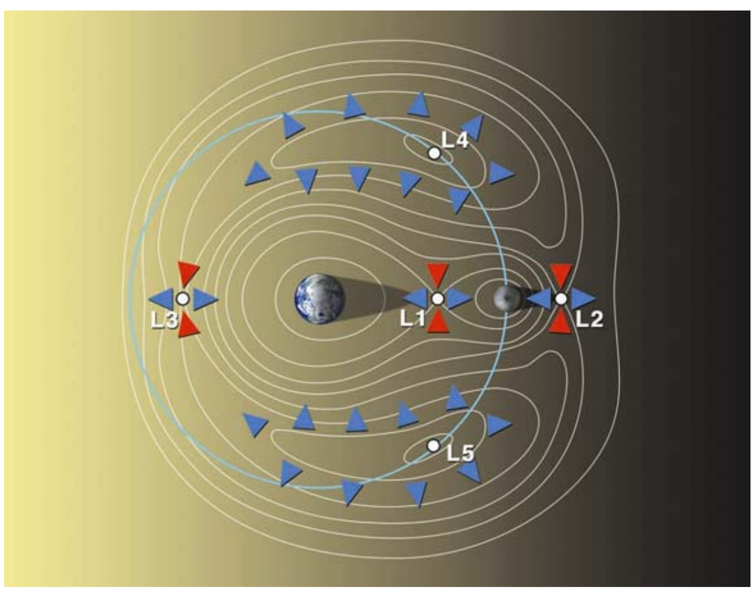


Figure 2. Lagrange Points in the Earth-Moon System (from Kring 2012)

The zone of stable equilibrium at the L1, L2, and L3 Lagrange points is very small and has been described as balancing a pencil on its tip where any perturbation will make it unstable. Therefore, spacecraft exploiting these points utilize a trajectory called halo orbits where the spacecraft orbit around the Lagrange gravitational equilibrium point in a circular path (Farquhar 1970). The L4 and L5 Lagrange points are very stable and can be visualized as an object at the bottom of a valley where any perturbation experienced results in the object moving back to the bottom of the valley. However, the L4 and L5 Lagrange points' distance to the Moon is too significant to be considered for lunar space power beaming applications. Research conducted on the use of L1 and L2 Lagrange points for the lunar South Pole coverage concluded that the use of two satellites in a single phased vertical butterfly orbit around the L1 libration point, as seen in Figure 3, provided continuous coverage of the lunar South Pole (Grebow 2008).

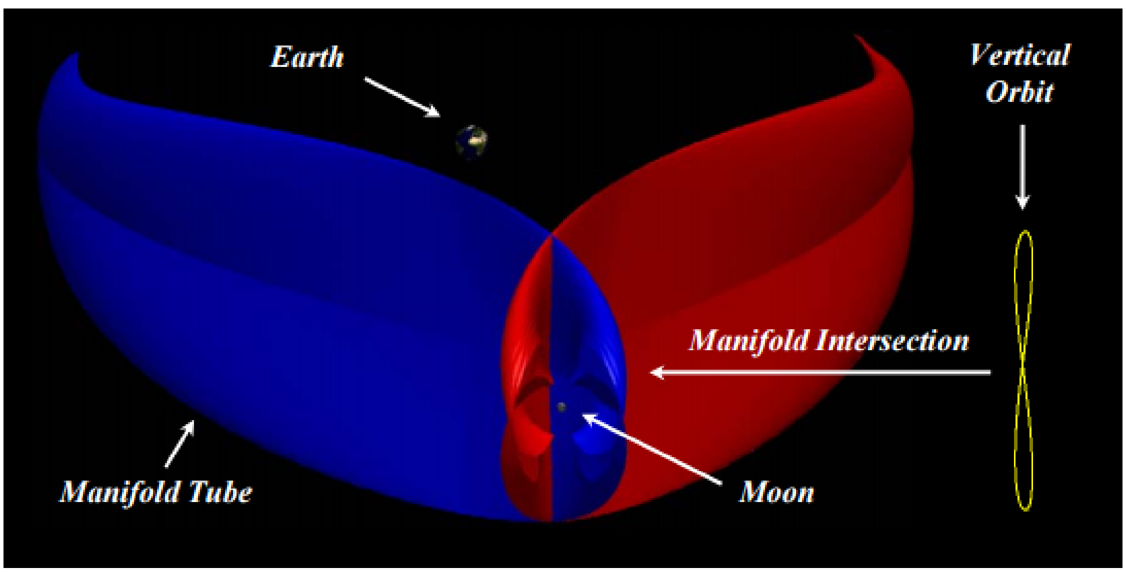


Figure 3. L1 Vertical Orbit with Intersecting Unstable (Red) and Stable (Blue) Manifolds from/to Earth and Moon Vicinities (from Grebow 2008)

B. LUNAR POWER GENERATION METHODS

The key question to answer on the feasibility of the lunar outpost is how to provide enough power at low mass and therefore low cost. Options include the use of surface photovoltaics with a complementary energy storage system, surface photovoltaics

at peaks of eternal light using cable transmission, or the use of nuclear power generation systems. The option of solar power satellites, which is the focus of this thesis, will be discussed in other sections. While each option is theoretically feasible, there are pros and cons with respect to cost/mass and/or safety.

The use of photovoltaic power generation systems directly on the Moon for polar regions has two important problems, which are the amount of available sunlight and the mass of the batteries. The amount of available sunlight at the Moon is determined by its orbit, with parts of the Moon receiving sunlight for approximately 14 terrestrial days (336 hours) followed by 14 terrestrial days of night. To overcome the problems of night time at a lunar base would require the use of batteries to act as an energy storage system. In the late 1980s, NASA conducted a study on the advanced photovoltaic power systems complemented by a storage system for lunar bases (Brinker 1988). NASA calculated the mass for both the photovoltaic and energy storage systems using a 100-kilowatt lunar base power requirement using figures of merit of power per unit mass (watts per kg, or W/kg) and available energy per unit mass (watt-hour per kilogram, or W-hr/kg). While the mass of the photovoltaic systems using a 300 W/kg figure of merit produced an exceptional total photovoltaic mass of 333 kg, the primary mass driver was the energy storage system when using an optimistic figure of merit of 1000 W-hr/kg. The final mass of the photovoltaic and energy storage system was 34.35 metric tons (mT) (34,350 kg) (Brinker 1988). While feasible, this mass would require several dedicated launches plus the lunar base still requires satellite coverage for communications and navigation. Therefore, this thesis assumes the use of a better architecture would be a combination of solar power satellite as the primary satellite payload with a secondary payload for communications and navigation.

The issue of surface photovoltaics that incur the problem of needing large energy storage mass can be overcome using peaks of eternal light. Peaks of eternal light are points on a body in the solar system that are in continuous sunlight. There are peaks of eternal light on the Moon due to the small axial tilt in the lunar orbit (Kruijj 2000). In a lunar power transfer feasibility study conducted in 2008, these peaks were assumed to be used with power cables as the transmission method (Popovic 2008). The lunar base

power requirement was 50 kW and the peaks of eternal light were considered within 2 km of the lunar outpost. A figure of merit for the specific mass per unit distance was 28.3 kg/1000 ft or 0.093 kg/m. The final results of the feasibility study for the use of peaks of eternal light using power cable transmission to lunar outposts was a total cable mass of 7500 kg (7.5 metric tons). This solution, compared to the surface photovoltaic and energy storage system, greatly reduces the amount of mass needed to be place on the Moon. The feasibility study identified problems other than the mass incurred when using power cables, including temperature sensitivity and solar flare induced transient effects. While these additional considerations may increase the mass of the peaks of eternal light solution, this option is highly competitive with the Solar Power Satellite system.

Recognizing the need for low weight power sources for space applications, in the 1980s, NASA, in conjunction with the Defense Advanced Research Projects Agency (DARPA), and the Department of Energy, began a program called the Space Reactor Project (SP-100) (Sovie 1987), as shown in Figure 4.

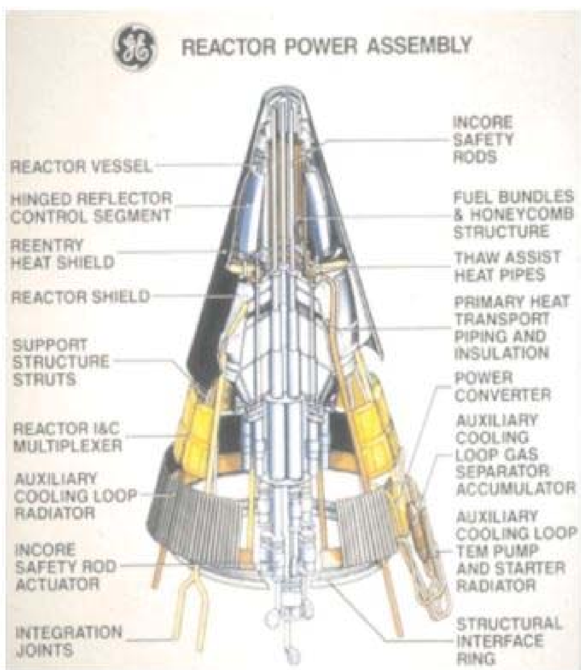


Figure 4. The SP-100 Space Nuclear Reactor Design (from Lior 2001)

The SP-100 reactor was designed to produce 100 kilowatts and be scalable to 1 megawatt with an initial total mass of 4600 kg or 4.6 metric tons, have a design life of 10 years, and provides an impressive figure of merit of system power to mass ratio of 217 W/kg (Marriott 1994). While the use of nuclear power for a lunar base is feasible, the primary issues are both political and safety related. As seen in the recent (2014–2015) launch failures of the Orbital Sciences Antares rocket, Russian Soyuz booster, and the SpaceX Falcon 9 in support of the International Space Station, there is still a high degree of technical risk associated with any launch (Achenbach 2015). The danger in using a nuclear reactor for lunar outposts is the risk of fallout from orbital debris should a launch failure occur. In addition, the nuclear reactor would have to land on the Moon, and should a landing failure occur could spread nuclear debris across a large range of the real estate that was intended for study. The use of nuclear reactors to power a lunar outpost, while feasible, carries associated political and safety risks too high to be considered given the available alternative of a solar power satellite.

C. WIRELESS POWER TRANSFER

Heinrich Hertz pioneered the technologies used today to transmit electricity though free space based on the fundamental work done by James Maxwell. Hertz demonstrated the possibility of wireless power transmission through his work on wireless communication using radio wave propagation for both transmitting and receiving high frequency electricity using a focusing technique by modification of antenna size and curvature. Nikola Tesla used the foundation set by Hertz in 1899 to achieve the first wireless power transfer of electricity. Tesla's efforts were largely unsuccessful even though physically possible because he used frequencies in the 150 kiloHertz (kHz) range and in excess of 100 million volts. The choice of such a high frequency and associated shorter wavelength was necessary to realize equipment of practical size since low frequency power transmission requires unreasonably large equipment. His unsuccessful attempts at wireless power transfer were due to the lack of technology able to handle the short wavelengths. The technology required to handle wireless power transmission of short wavelengths, even for small amounts of power, was not available for another 40 years (Brown 1984). It was with the invention of the magnetron and klystron, both

developed during World War II, that the use of microwaves became available for effective transmission of energy. However, the application of microwaves for wireless power transfer would not be completed until the 1960s, when William Brown used microwave transmission to power a tethered helicopter (Brown 1984). Since these early efforts by Hertz, Tesla, and Brown, the technology for wireless power transfer, specifically the use of microwaves, has advanced significantly. Recently, in March of 2015, Mitsubishi Heavy Industries with the sponsorship of the Japan Aerospace Exploration Agency (JAXA) sent 10 kilowatts of power through the air over 500 meters to a receiver with high accuracy (Mitsubishi 2015).

The idea of using light produced by lasers for wireless power transfer is more recent since the theoretical analysis for a laser was completed in the 1950s and a functioning laser was presented in the 1960s (Gould 1959). The use of light produced by lasers for wireless power transfer consequently requires the use of specialized photovoltaic cells to receive and convert the energy. The development and use of photovoltaic cells also goes back to 1960s when they were first mounted on spacecraft to complement the batteries and extend the spacecraft's operational life. Research and advances are continuing for specialized photovoltaics for laser application (Fast 2011).

Microwaves are the default choice for wireless power transfer and a significant amount of scientific research, development, and testing has been conducted to advance the use of this frequency range. Microwaves are defined in a broad sense as the ultra-high frequency (UHF) to the extremely high frequency (EHF) part of the electromagnetic spectrum with frequencies between 300 MHz and 300 GHz with corresponding wavelengths of one meter to one millimeter. In the 1970s, NASA in conjunction with the Department of Energy conducted tests using the EHF portion of the electromagnetic spectrum for wireless power transfer which would become the standard approach for microwave wireless power transfer (Brown 1984). The standard approach included the use of a parabolic transmitter pointed at a receiving array. The receiving array used half-wave dipole antennas, which themselves required extensive development, and became known as rectennas (Brown 1984). The fundamental equation used to describe how the

selected frequency is proportional to the half-wave dipole size and consequently the size of the rectenna is given by:

$$\lambda = \frac{c}{f} \tag{2}$$

where λ is the wavelength, c is the speed of light in a vacuum, and f is the frequency (Gordon 1993). The NASA tests used the 2.5 GHz frequency to transmit 30 kW of power over 1600 meters using an approximately 27 square meter rectenna with an impressive direct current (dc) to dc efficiency of 54% (Brown 1984). One of the many difficulties of wireless power transfer using microwave frequencies is the power lost due to atmospheric effects. Luckily, for the application of microwave power beaming from a solar power satellite to a lunar outpost, there are no atmospheric effects, and this loss does not need to be considered.

Applying the fundamental equation (2) and understanding the proportionality of the rectenna size and frequency, the goal of recent research has been to increase the frequency and therefore utilize smaller wavelengths to achieve smaller transmitter and receiver antenna sizes. If this thesis were purely theoretical, then the best microwave frequency to use would be 300 GHz, as this results in the smallest antennas; however, as mentioned in the introduction, the goal of this research is to choose relevant and practical parameters and there are no antennas that can transmit and receive this frequency. Recent research using a two-dimensional slot antenna produced by a well-established manufacturing process for silicon chips, called photolithography, has created antennas that can receive frequencies between 70 GHz and 150 GHz (Marzwell 2008). While the slot antenna can handle frequencies between 70 GHz and 150 GHz, it has been optimized for 94 GHz and has a radio frequency (RF) to direct current (dc) conversion efficiency of 93% and a collection efficiency of 72% with power densities as large as 1 Watt per square centimeter (Watt/cm^2) (Marzwell 2008). Lacking from this research was an important figure of merit needed to understand this technology's impact on the architecture which was the mass per unit area. A substitute mass per unit area of 0.16 kilograms per meter squared (kg/m^2) was used from research on the development of rectenna technologies for the 2.45 GHz and 20 GHz frequency (Brown 1987).

In addition to technology improvements in microwave receiver technology to increase the selected frequency to drive down antenna sizes, research has been conducted to increase the size of antennas on orbit. In 2010, Terrestar produced and launched the largest commercial antenna on the SkyTerra-1 satellite with a 22-meter antenna reflector (Amos 2010). The SkyTerra-1 satellite used large deployable reflectors that were tested up to frequency ranges of 15 GHz (Datashvilli 2010). The need for large antenna sizes for microwave power beaming is driven by the divergence of the RF beam by diffraction and this divergence increases as the distance between the transmitter and receiver increases. The equation which describes the divergence of the RF beam is given by:

$$\frac{D_t D_r}{\lambda x} = 2.44 \tag{3}$$

Equation (3) describes the relationship as a function of the distance between the transmitter and receiver, defined as x, the transmitter antenna diameter size D_t, the receiver antenna diameter size D_r, and the wavelength λ.

Independent of the distance between the transmitter and receiver, and the power intensity of the beam, the transmitted power in the main lobe of the RF beam will be 84%. The power intensity of an RF beam and specifically the power in the main lobe at the receiver is given by the equation:

$$I_0 = \frac{\pi P_t}{4}\left(\frac{D_t}{\lambda x}\right)^2 \tag{4}$$

Equation (4) describes the relationship of the power intensity, I_0, within the main lobe of the beam at the receiver relative to the amount of transmitted power, P_t, and as a function of the distance and wavelength (Potter 2009). Given the concepts above on microwave power transmission these parameters can be used to understand their impact on the solar power satellite architecture.

The laser has a long history of both theoretical and material origin, when Max Plank formulated the underlying idea that light was just another form of electromagnetic radiation and received the Nobel Prize for it 1918 (Hecht 1992). Albert Einstein used Max Plank's law of radiation and firmly established the theoretical fundamentals in his

paper called “On the Quantum Theory of Radiation” (Einstein 1917) which outlined how the absorption and stimulated emission of electromagnetic radiation could work. Building on yet more theoretical work done by several scientists, in May of 1960, Theodore Maiman created the first operating laser at Hughes Research Laboratory in California (Maiman 1960). At the simplest level, a powered laser acts as a light source in which the direction and wavelength of the light can be controlled. At a more complicated level, a laser is the result of electrons moving to higher atomic energy levels when given energy by light or heat, the resulting higher atomic energy level electron will eventually drop back down to a lower energy state and during this transition will release a photon. If the electron drop to a lower energy level is random, then the emission of a photon is called spontaneous emission. When it is controlled through the use of an external electromagnetic field and uses a frequency unique to the specific material to transition electrons between two atomic states, it is called stimulation emission. The culmination of this theoretical and material research has been a technology whose applications range from optical disk drives, fiber optic cables, laser surgery, welding, and the applications keep growing (Rose 2010).

To use a laser in a solar power satellite requires the understanding of three parameters. The first parameter is the total amount of power that a laser is required to generate for a lunar outpost. The second parameter is the electrical to optical conversion efficiency which varies considerably and is contingent upon on the method used to generate and control the laser. The third parameter is the power lost to beam spreading for a given distance. Since the total amount of power provided by the laser to the lunar outpost is determined by the outpost requirements and the electrical-to-optical conversion efficiency depends on the type of laser technology used, the primary understanding needed for laser power transmission is the power lost due to beam spreading. Luckily for the application of a laser solar power satellite powering a lunar outpost, the effect of atmospheric transmittance can be ignored since the Moon does not have an atmosphere to attenuate the laser.

Since the laser will propagate through a vacuum, it will be treated as a Gaussian beam. A Gaussian beam is used in optics when a beam of electromagnetic radiation has

an electric field and intensity profiles in the transverse plane that can be approximated by Gaussian functions (Svelto 1976). From the Gaussian functions the beam width or spot size can be determined by calculating the beam radius. The beam radius, w_z, will vary along the beam axis but will be at a minimum at one point along the axis, called w_o, when the beam is at its smallest area. This relationship is illustrated in Figure 5 and defined by the following equations:

$$w(z) = w_0\sqrt{1+\left(\frac{z}{z_R}\right)^2}$$
$$z_R = \frac{\pi w_0^2}{\lambda} \tag{5}$$

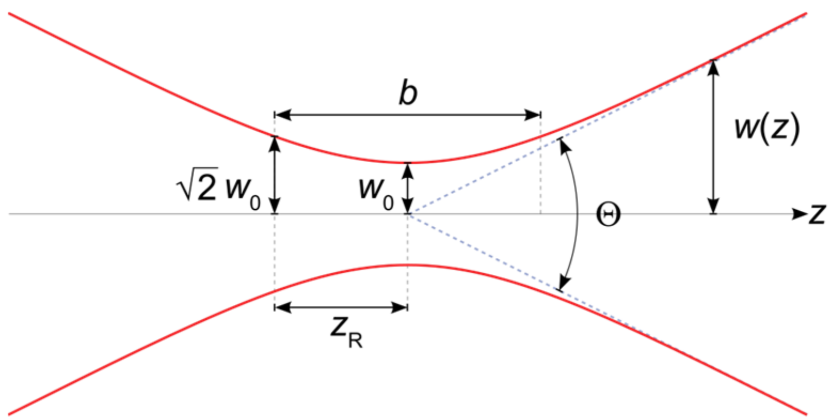


Figure 5. Gaussian Beam Waist (from Bob 2009)

Equation (5) describes the beam radius as a function of the beam waist at its smallest point and depends on the wavelength used and the distance the laser is traveling. From equation (5) the wavelength used will have an impact on the beam waist and decreasing the wavelength will reduce the diffraction of the beam. Beam expanders, which are lenses, can also be used to control the beam waist and reduce the diffraction of the beam. Finally, using the area of a circle in Equation (6) the spot size area at the lunar outpost receiver can be determined.

$$A = \pi w(z)^2 \tag{6}$$

Ultimately, the goal of these equations and their application to the solar power satellite architecture is to understand how the laser beam will spread out over the desired distance and to determine the size of both the solar power satellite transmitter and lunar outpost receiver.

With the understanding of laser beam propagation and its dependence on wavelength, the choice of laser technology can be determined. Laser wireless power transfer has a significant degree of interdependence between the laser transmitter and receiver for a given laser material. The choice of laser technology will generate a particular wavelength that the receiver must be able to handle. The wavelength range for different laser materials is shown in Figure 6.

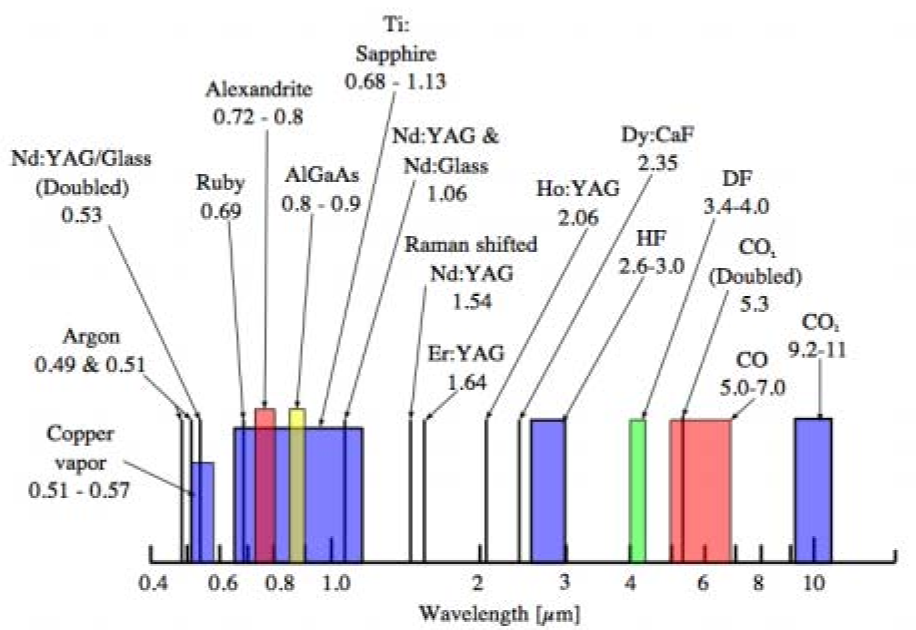


Figure 6. Spectral Lines of a Variety of Laser Materials (from Command 2005)

For this thesis on the feasibility of a solar-powered satellite providing power to a lunar outpost, a choice of laser transmitter and associated wavelength, as well as receiver technology must be made. Two laser technologies were considered to understand their

impacts on the solar power satellite architecture: the stretched lens array (SLA) and the Vertical Multi-Junction (VMJ) photovoltaic cell.

Since the 1980s, several agencies including NASA and ENTECH have worked on a photovoltaic concentrator system for both solar and laser receivers called the stretched lens array (Piszczor 2006). The SLA was specifically designed with the idea of using this technology in the dark craters at the lunar poles (O'Neill 2006). The SLA uses a straight line array of arched lens to concentrate and focus light up to 8.5X its original intensity on triple junction photovoltaic cells as seen in Figure 7 and Figure 8.

Figure 7. Stretched Lens Array Prototype (from O'Neill 2006)

Figure 8. SLA on ATK Space's SquareRigger Platform (from O'Neill 2006)

While both the solar and laser versions of this technology will be used for the architecture analysis, the real importance to this thesis are the figures of merit for the

laser version of the SLA. During hardware tests and using a laser wavelength of 805 nm, the SLA had an operational array efficiency of 41.4% for collection and conversion, with a power density of 690 W/m^2 (0.069 W/cm^2) and a specific mass to area density of 0.86 kg/m^2 (O'Neill 2006).

Another recent receiver technology for laser power transmission is the VMJ. The VMJ will act as the lunar outpost receiver and has the ability to continuously convert a high-intensity laser beam into electricity. The VMJ's unique ability is based on photovoltaic technology using a serially-connected array of small silicon junction cells that have the ability to directly convert high-intensity optical energy into electrical energy (Goradia 1977). Normal photovoltaic cells convert sunlight at 1 astronomical unit (a.u.) which is equivalent to 1370 W/m^2 from optical energy to electrical energy. However, the laser power transmission will be several orders of magnitude greater than one sun's worth of irradiance. Therefore, for the application of laser wireless power transmission it is necessary to use a receiver that can handle high-intensity energy. Recent research conducted in 2011 on VMJ technology has created a receiver that can convert optical energy when flashed at solar irradiance levels as high as 2500 suns and equal to 211 W/cm^2 producing over 40 watts of electrical output which is a conversion efficiency of 23% using a laser wavelength of 980 nm (Fast 2011). Figure 9 shows a schematic view of a silicon VMJ photovoltaic cell.

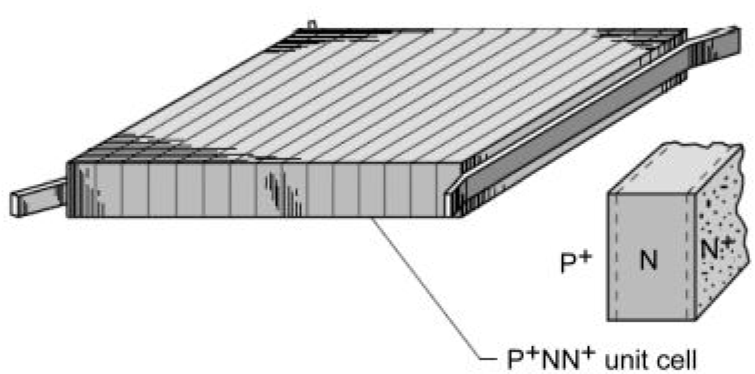


Figure 9. A 40-Junction Silicon VMJ Photovoltaic Cell (from Raible 2011)

Recent experimental hardware results using a single photovoltaic VMJ cell under high intensity continuous-wave laser irradiance produced 13.6 W/cm^2 with an optical to electrical conversion efficiency of 24% using a laser wavelength of 976 nm (Raible 2011). These results lead to a potential VMJ receiver size of 1 m^2 which can convert approximately 136 kW of energy (Raible 2011). Figure 10 shows optical bench testing for a VMJ prototype.

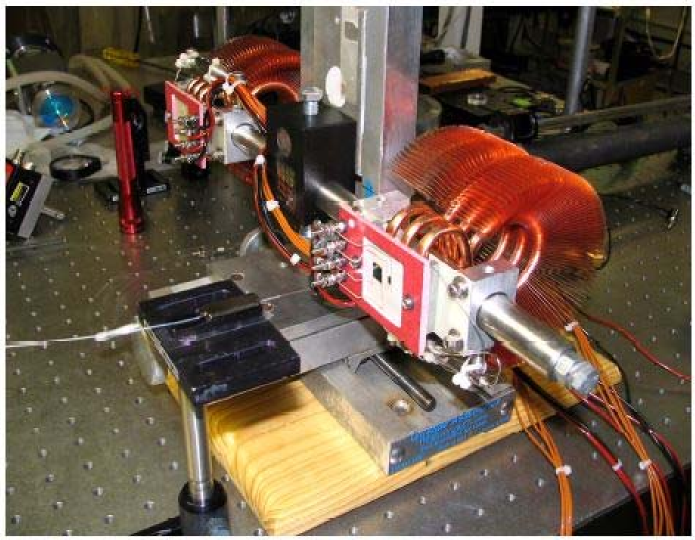


Figure 10. Optical Bench Setup during the Laser Test (from Fast 2011)

Missing from the papers done by Fast and Raible was a key figure of merit for VMJ technology which was the specific mass to area density (kg/m^2). A research paper conducted on the trade space for photovoltaic cells by Fatemi in 2000 gave a realistic figure of merit of triple junction solar cells of 0.85 kg/m^2 (Fatemi 2000).

D. SOLAR POWER SATELLITE HISTORY AND ARCHITECTURES

Since the original idea of a solar power satellite by Dr. Glaser in 1968 there have been numerous studies trying to advance the concept to practical application. While most solar power satellite concepts are feasible, the amount of mass required to put a solar power satellite into orbit; therefore, the cost of each concept has proved unreasonable. However, the trend line for the amount of power per unit mass for each successive solar

power satellite concept has been in the direction of reducing mass and increasing power. The application of this technology is inevitable. The first data point for the key figure of merit of total system power per unit mass is from Dr. Glaser's initial concept. Dr. Glaser used a satellite with a solar conversion area of 270 km^2 (or a satellite with a diameter of 11.5 miles) with just a solar conversion area estimated mass of 1.8 x 10^8 lbs (or 81.5 x 10^6 kg) using a photovoltaic conversion efficiency of 24% to satisfy the electrical demand of the northeastern United States with an estimated annual power consumption in 1980 of 0.50 x 10^{12} kW-hrs (Glaser 1968). This results in a satellite solar conversion power per unit mass of 0.7 kW/kg. Once other satellite subsystem masses are included, the total satellite power per unit mass would likely drop significantly below 0.5 kW/kg. An important reference point for economic viability for terrestrial solar power satellites is above 1 kW/kg for good returns on investment or above 0.5 kW/kg total system power per unit mass for competitive economic viability (Mankins 2002).

From 1976 to 1980, the Department of Energy in conjunction with NASA conducted the first major study of a solar power satellite architecture, which became known as the 1979 Reference Design (NASA 1978), as illustrated in Figure 11.

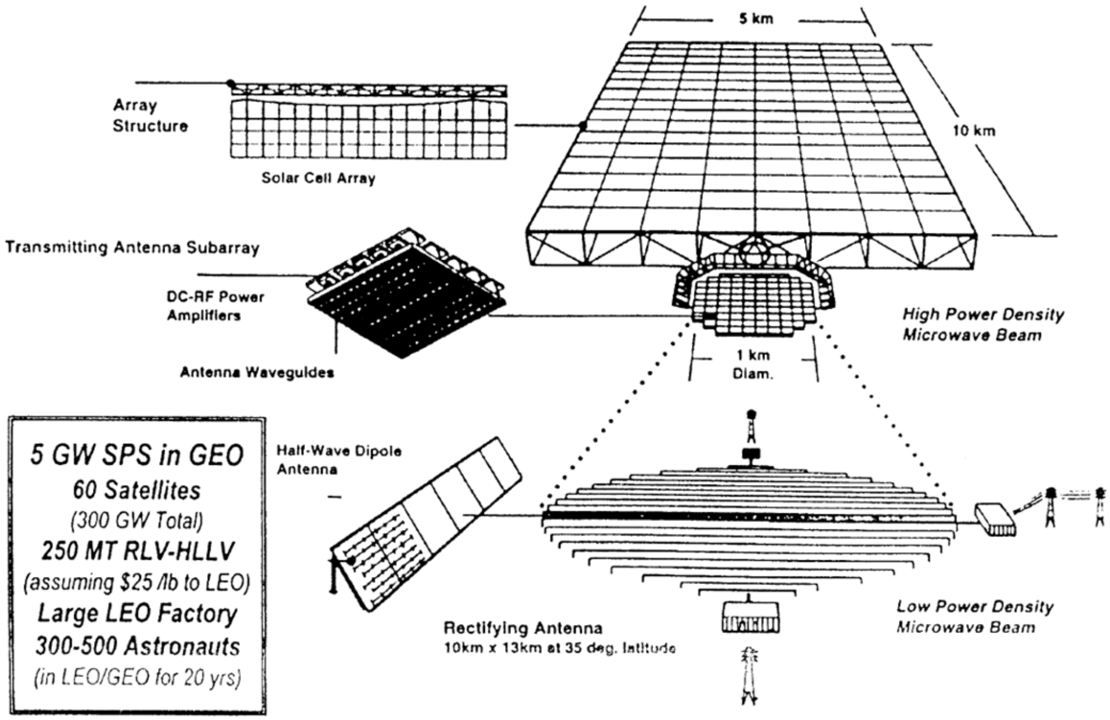


Figure 11. 1979 SPS Reference System Concept: 5 GW Power Output, Geostationary Earth Orbit-Based Systems (from Mankins 2002)

The 1979 Reference System concept produced 5 GW of electrical power using gallium-aluminum-arsenide (GaAlAs) photovoltaics and utilized the 2.45 GHz wavelength to ground receiving rectennas with a total spacecraft system mass of 34 x 10^6 kg (NASA 1978). This resulted in a total system power per unit mass of 0.15 kW/kg. The 1979 reference designed was composed of large erected structures using dozens of launch vehicles and a construction facility in low earth orbit utilizing hundreds of astronauts. Ultimately, the U.S. National Research Council and the now defunct Congressional Office of Technology Assessment concluded that while this concept was feasible, the cost was unachievable. It had a cost estimate of $250 billion in 1996 dollars (Mankins 1997).

From 1995 to 1997 NASA conducted an exhaustive study and re-examination of technologies and system concepts for solar power satellites called the “Fresh Look” study (Mankins 1997). Out of this study was a new solar power satellite concept called the

SunTower which is a highly-modularized and gravity-gradient stabilized architecture as shown in Figure 12.

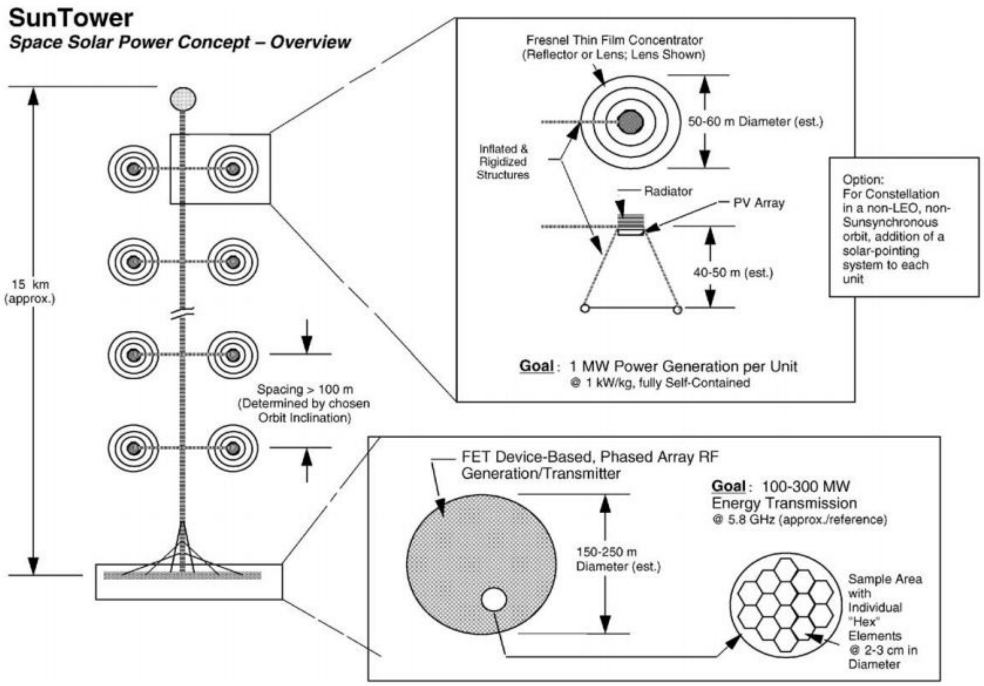


Figure 12. The "SunTower" Solar Power Satellite Concept (from Mankins 2002)

The SunTower SPS concept used the 5.8 GHz frequency with an initial 1000 km sun-synchronous orbit paired with a ground receiver sized to handle between 100 to 400 megawatts. The cost for a 250 MW SunTower platform was estimated at $15 billion dollars and designed with a total system power per unit mass of 1 kW/kg (Mankins 2002).

After the 1997 Fresh Look study concluded that solar power satellite concepts were reaching technical maturity and economic viability, NASA began the Space Solar Power (SSP) Exploratory Research and Technology (SERT) program which ran from 1999 to 2000. The goal of the SERT program was to define a strategic technology roadmap to deliver large megawatt systems using wireless power transmission for both government science missions and commercial markets (Howell 2000). Out of the SERT

program came the solar power satellite concept called the Integrated Symmetrical Concentrator (ISC) as illustrated in Figure 13.

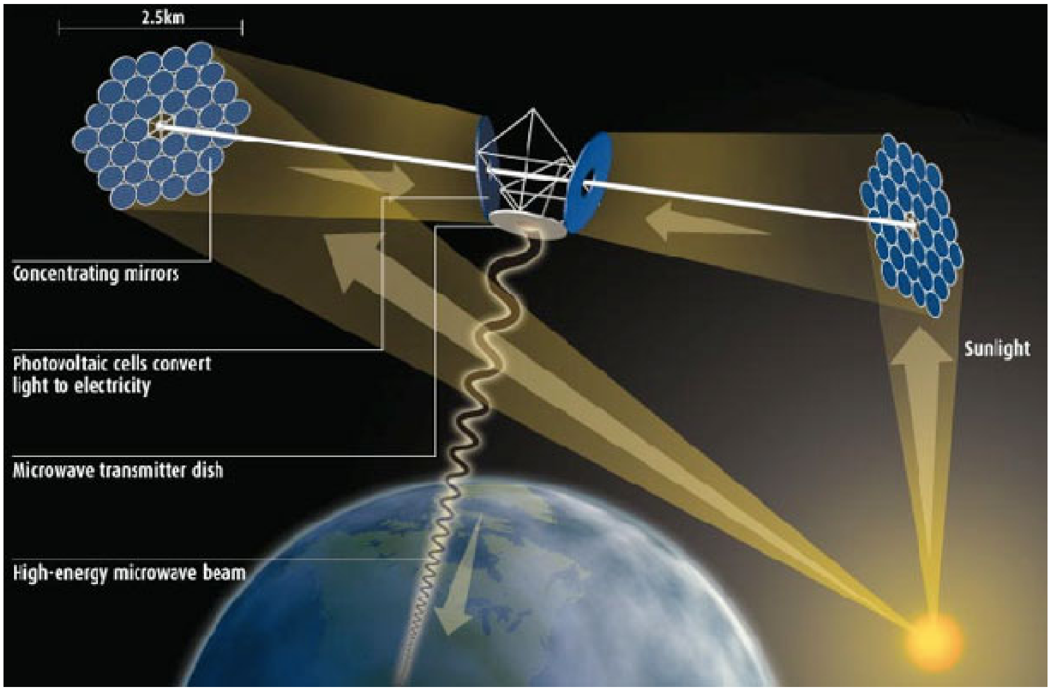


Figure 13. Integrated Symmetrical Concentrator SPS Concept
(from Belvin 2010)

The ICS was modeled for a full scale 1.2 GW delivered to the grid system with a wireless power transmission efficiency of 37% from geostationary orbit using a 5.8 GHz microwave frequency. The ICS had with a 7450 meter diameter rectenna and the resulting mass was 17,076 kg or 17 metric tons (Carrington 2000). The ISC concept was designed with a total system power per unit mass figure of merit of 1 kW/kg.

Completed in 2011, a group of academics studied the impact space solar power might have to satisfy the global energy demand in the 21st century called the First International Assessment of Space Solar Power (Mankins 2011). Their goal was to assess the technology readiness level and risks associated with various solar power satellite concepts and to build a technology roadmap that might lead to the first demonstration and operational solar power satellites. This study evaluated numerous concepts and concluded that three concepts were highly promising, that there were no fundamental technical

barriers, and that the primary issue was the economic viability of solar power satellites. Information from this study was included in the thesis for two reasons; it provided a suitable functional architecture and included an exceptional systems analysis methodology using figures of merit for elements in the functional architecture. A modified version of this functional architecture and figure of merit methodology will be used in Chapter III of this thesis.

To evaluate and compare different solar power satellite concepts it was necessary to define a common functional architecture grouped into primary SPS platform systems, secondary SPS platform systems, ground systems, and supporting systems/infrastructure elements. The generic functional architecture used in the International Study can be seen in Figure 14. As seen in Figure 14, the solar power satellite was broken into five elements, to include the solar power generation system, the power management and distribution system, the thermal management system, wireless power transmitter, and the wireless power transmission receiver. The generic functional breakdown also included the launch system, in-space transport system, and ground assembly and operations infrastructure.

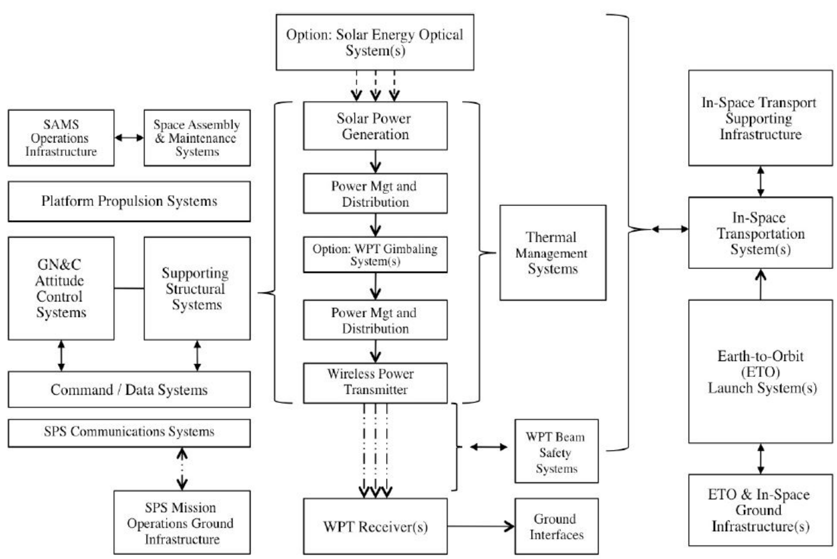


Figure 14. Generic SPS Functional Architecture (from Mankins 2011)

Given this functional architecture, the systems analysis methodology identified figures of merit for each element in the system and the interrelationship between the elements' figures of merit, as seen in Figure 15. In order to represent just the solar power satellite, the study used 31 figures of merit, as seen in Figure 16.

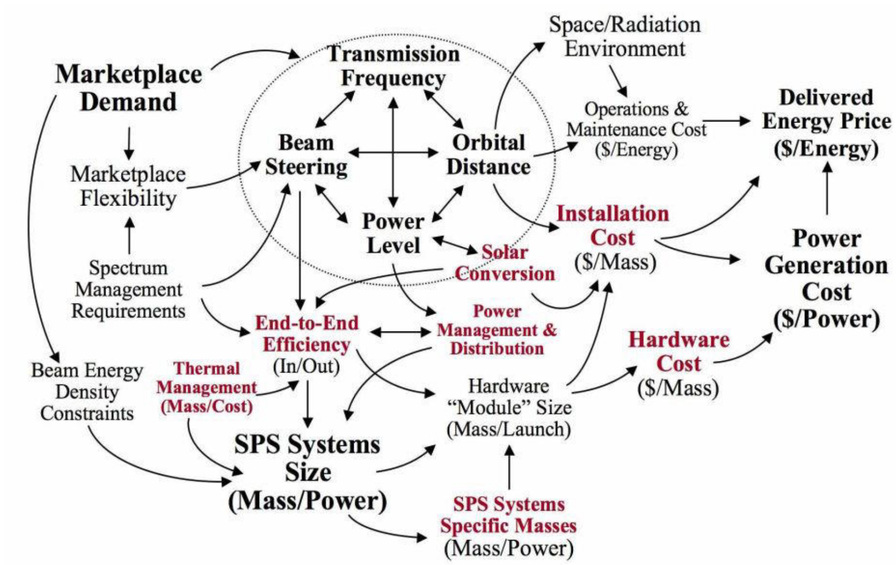


Figure 15. Interrelationship of SPS Element Figures of Merit (from Mankins 2011)

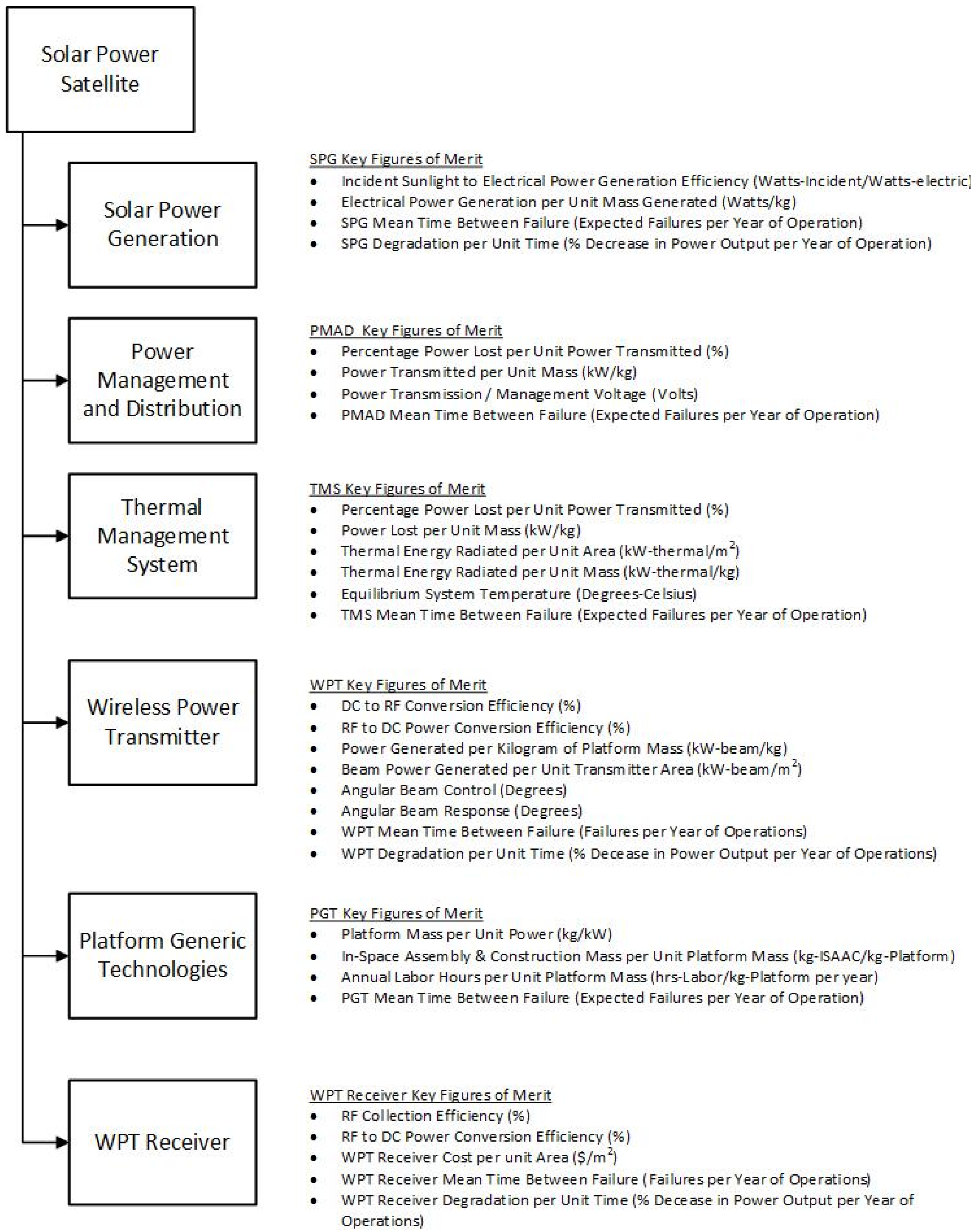


Figure 16. Space Solar-Power Selected Figures of Merit (after Mankins 2011)

Under the NASA Innovative Advanced Concept Program conducted from 2011 to 2012, an innovative solar power satellite concept was developed that mitigated many of

the economic viability problems identified in the First International Assessment of Space Solar Power. The primary editor of the international assessment and principal investigator for SPS-ALPHA report was John C. Mankins. The Solar Power Satellite via Arbitrarily Large Phased Array (SPS-ALPHA) project's goals were to do an analytical proof of concept backed by an end-to-end systems analysis, identify the key technology challenges using figures of merit, and finally conduct an economic viability of the concept (Mankins 2012). The SPS-ALPHA concepts can be seen in the right of Figure 17 with earlier solar power satellite concepts.

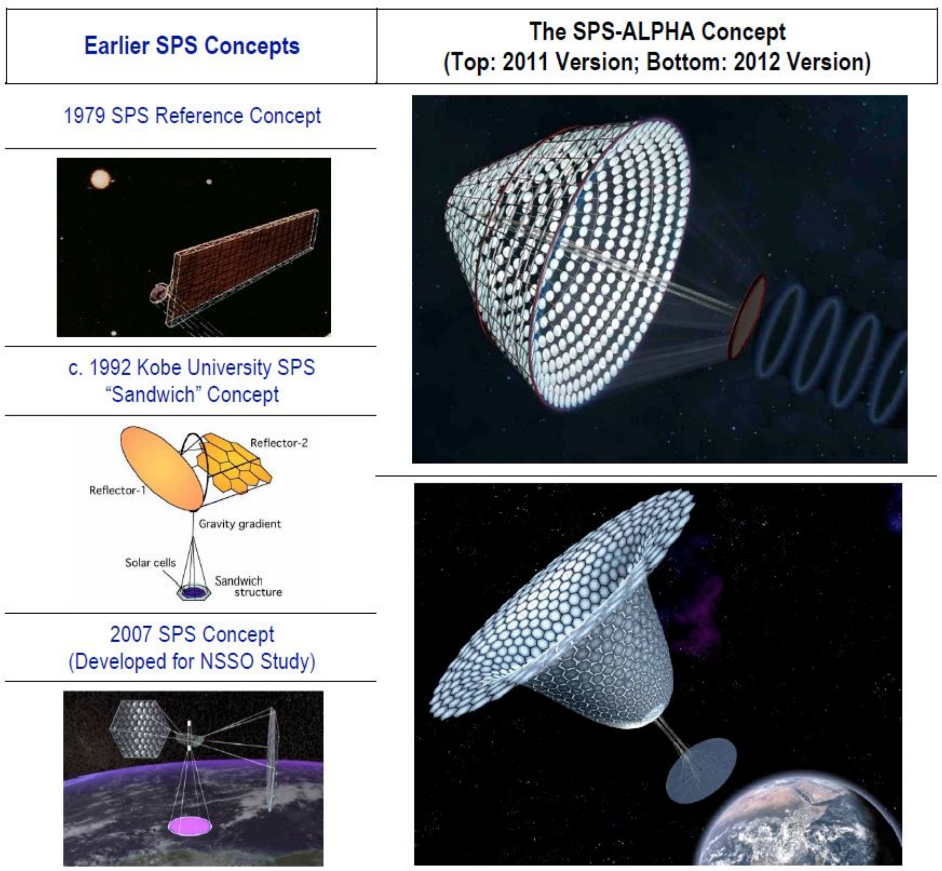


Figure 17. Solar Power Satellite Concepts and Two Versions of the New SPS-ALPHA Concept (from Mankins 2012)

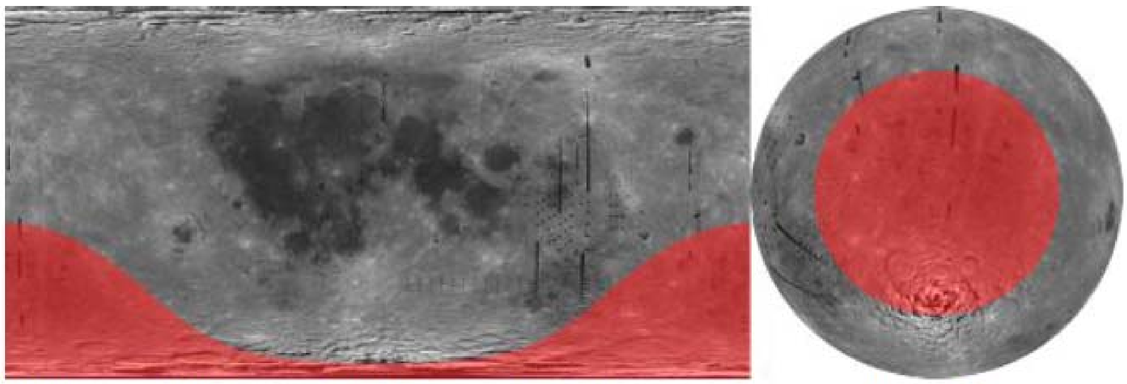
The SPS-ALPHA concept used the functional architecture and system analysis methodology of figures of merit from the international assessment to reach their conclusions. To limit the mass impact, the SPS-ALPHA concept used a very modular

design due to recent technological advances to deal with the power management and thermal management issues inherent in large solar power satellite designs. Two solar power satellite concepts were chosen for data points for the key figure of merit of total system power per unit mass. The first SPS-ALPHA concept was a 200 kW delivered to the grid low earth orbit system using the 2.45 GHz frequency and an end-to-end efficiency of 10% and final mass of 16,768 kg or 16.8 metric tons (Mankins 2012). The total system power per unit mass was 0.12 kW/kg. The largest and most advanced version of the SPS-ALPHA concept was for a 2 GW delivered to the grid geostationary solar power satellite using the 2.45 GHz frequency with an end-to-end efficiency of 10% with a total mass of 25,260,800 kg or approximately 25,260 metric tons. This results in a total system power per unit mass of 0.79 kW/kg.

E. LUNAR EXPLORATION RESEARCH

The National Research Council (NRC), in 2007, at the request of NASA, produced a report called "The Scientific Context for Exploration of the Moon" (Board 2007) for a lunar exploration program that provided global access to the surface with a combined human and robotic architecture. The underlying premise of the NRC report which was for a global, high mobility, and long duration mission, was a significant departure from prior lunar scientific missions that had been limited to the nearside equatorial region for short duration trips. It is important to note that most of the Moon has never been explored either at the far side, along the western limb, or in the polar regions. The NRC report captured the scope of the science to be done at the Moon within eight key scientific concepts with 35 prioritized scientific goals with four to five goals per scientific concept (Board 2007). The fundamental question after completing the NRC report was where on the lunar surface should a spacecraft and astronaut land to satisfy as many of the scientific goals as possible. To address this question, NASA created the Lunar and Planetary Institute (LPI) and Johnson Space Center (JSC) Center for Lunar Science and Exploration to conduct a landing site survey using the NRC science concepts as their selection criteria (Kring 2012). For five years, eight teams, each assigned a scientific concept, evaluated the lunar surface to create landing sites where the scientific goals for each concept could be addressed. The results were a global assessment and a

comprehensive landing site study which recommended hundreds of locations to land to satisfy specific scientific goals with several sites being uniquely scientifically rich. However, one landing site stood out. That site was the Schrödinger basin within the South Pole-Aitkin Basin (SPA Basin) on the lunar far side as seen in Figure 18 and Figure 19.



The location of SPA, first on the left in a cylindrical projection centered on the central nearside, and then on the right in an orthographic projection centered on SPA.

Figure 18. Location of the SPA Basin (after Kring 2012)

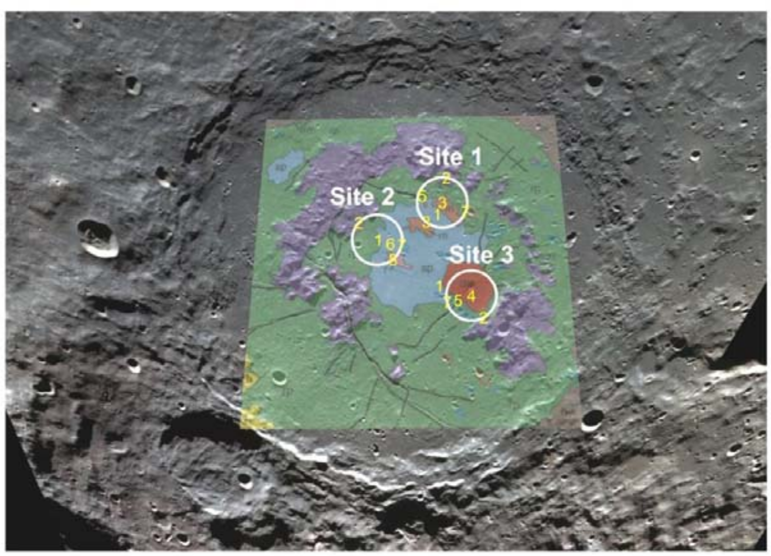


Figure 19. The Three Landing Sites and Corresponding 10 km (20 km return trip) EVA Radius for Schrödinger (from Kring 2012)

It was determined that at this location the first and second priorities of the NRC report could be accomplished as well many of the scientific goals of the eight scientific concepts within the 2,500 km diameter of the South Pole-Aitkin basin. Another significant scientifically rich location was the Amundsen crater near the South Pole which was singled out due to the polar volatiles research that could be accomplished. Given the selection of the SPA Basin, a feasibility assessment of all science concepts within the SPA Basin was added to the Landing Site report as a distinct section.

F. ADDITIONAL FIGURES OF MERIT

In order to compare different solar power satellite concepts a set of common figures of merit are required which include the solar power satellite cost per kilogram, power per unit mass, and lunar cost per unit mass which is the cost to place 1 kilogram on the lunar surface. The solar power satellite cost per kilogram in both the First International Assessment on Space Solar Power and the SPS-ALPHA concept used a figure of merit of $100,000 per kg (Mankins 2011). As a point of comparison the block-buy of the fifth and sixth Advanced Extremely High Frequency (AEHF) which is a highly complex protected communications satellite was $1.93 billion (Ferster 2012). AEHF satellites have a mass of 6,168 kg, which results in a satellite cost per kilogram of approximately $156,000 (Krebs 2015). The choice of a $100,000 per kilogram used in this thesis for a solar power satellite seems reasonable since the design of a solar power satellite is expected to be simpler than an extremely complex protected communications satellite. The next figure of merit required is the total system power per satellite mass (kW/kg). This parameter is especially helpful in understanding how much power can be produced for each kilogram of satellite mass, a high-efficiency and low mass solar power satellite with have a figure of merit greater than 200 watts per SPS kilogram (0.2 kW/kg) (Mankins 2011). The final common figure of merit is the lunar cost per unit kilogram which is the cost to place one kilogram safety on the lunar surface. From the Parametric Model of a Lunar Base for Mass and Cost Estimates by Peter Eckart stated that the cost to land mass on the Moon is on the order of five to 10 times higher than the cost per unit mass to LEO (Eckart 1996). The 1996 cost per unit mass for 1 kg to LEO was $10,000 (Eckart 1996). A lunar cost per unit kilogram of $100,000 was used (Eckart 1996). In

addition to the common figures of merit, the solar power generation system will need two figures of merit for collection efficiency and DC to microwave/laser efficiency. The solar version of the SLA will be used as the photovoltaic technology on the SPS for both wireless power transmission technologies which has an operational array efficiency of 25% (O'Neill 2006). The SLA collection efficiency will turn approximately 25% of the incoming sunlight into direct current (DC) which will be converted to microwave and laser power with a conversion efficiency of 78% (Tanwar 2013) and 80% (Summerer 2009), respectively.

G. LUNAR OUTPOST POWER REQUIREMENT

A lunar outpost for human exploration of the Moon with long duration and high mobility missions will require a significant amount of power. As mentioned in the introduction the International Space Station is a useful reference point as six crew members require approximately 100 kW of energy (Wells 2009). In addition to the ISS reference point, this power problem has been studied extensively in the *Lunar Base Handbook* where even a minimal permanent base requires 50 kW and up to one megawatt for large bases (Eckart 1999). The challenge is overcoming the two-week lunar night when using surface photovoltaics in combination with an energy storage system. A minimal base at a polar site with a 50 kW power requirement requires an approximate 11,000 kg regenerative fuel cell storage system, and an equatorial lunar base has an energy storage mass of 17,000 kg (Kare 2004). The mass estimates by Kare match the numbers stated early by Brinker at 34,350 kg for a 100 kW lunar base. In the following section, this thesis will assume a minimum power requirement for a lunar outpost of 100 kW and will vary the power requirement to understand the impact on the solar power satellite system mass and cost. A research paper on the comparison of different power generation methods showed that laser based approaches exceed other methods when power requirements are greater than 300 kW (Bozek 1993).

H. LAUNCH VEHICLES

Two launch vehicles were chosen to understand their impact on the total system cost and on the upper boundaries of the solar power satellite mass. The two launch

vehicles selected were the Space Launch System (SLS) and Falcon 9 Heavy. The SLS is a heavy launch vehicle designed by NASA and meant to be upgraded over time. The first SLS version called Block 1, will be capable of putting 70,000 kg into low Earth orbit or 25,000 kg in a trans-lunar injection (TLI) orbit for an estimated cost of \$600 million dollars (Skran 2015). In comparison, the SpaceX Falcon 9 Heavy, a commercially developed heavy launch vehicle, has an estimated low Earth orbit payload of 53,000 kg or a 13,200 kg trans-lunar injection orbit with an estimated price of \$158 million dollars (Skran 2015).

With the mass of the launch vehicles at the trans-lunar injection point a final propulsive maneuver will need to be executed which will decrease the amount of mass available for the solar power satellite payload. The ideal rocket equation (7) can be used to determine the mass impact of changing the orbit from a trans-lunar injection to a low lunar orbit.

$$\Delta v = I_{sp} g_{earth} \ln\left(\frac{m_{initial}}{m_{final}}\right) \tag{7}$$

where the Δv (delta-v) is the change in velocity required to reach a new orbit, I_{sp} is the specific impulse which is a measure of the rockets efficiency, g_{earth} is the gravitational constant of the Earth, $m_{initial}$ is the mass at the trans-lunar injection point, and m_{final} is the mass after the propulsive maneuver to enter lunar orbit and subsequently the available mass for the solar power satellite.

The delta-v required to change an orbit from trans-lunar injection to a low lunar orbit is 0.8 km/s (Biesbroek 2000). The gravitational constant of the Earth is 9.807 m/s^2. The specific impulse is determined by the space propulsive technology used. Two propulsive technologies were chosen to understand their impacts to the solar power satellite available mass. The first propulsive technology used was a traditional chemical rocket with an average specific impulse 225 seconds (Buchheim 2007). The second propulsive technology selected uses hall current thrusters - a type of ion engine. Hall current thrusters (HCT) have an outstanding specific impulse and greater flexibility than traditional chemical rockets; however, their thrust is small and applied over a longer time

span. If the goal is to reach the destination quickly, then traditional chemical rockets are required. Recent data from the use of hall current thrusters on the AEHF satellite showed a typical specific impulse used of approximately 2100 seconds (Welander 2001). When applying the rocket equation to the TLI mas of the Falcon 9 Heavy and solving for mass final using traditional chemical propulsion, the final mass resulted in a decrease to 9,186 kg remaining for the payload, and with hall current thrusters, the final mass was 12,697 kg. When applying the rocket equation to the TLI mass of the SLS Block 1 and solving for mass final using traditional chemical propulsion, the final mass resulted in a decrease to 17,397 kg remaining for the payload, and with hall current thrusters, the final mass was 24,047 kg. These results show that a low energy and long duration transfer will be required for inefficient and therefore high mass solar power satellites.

III. SYSTEM ARCHITECTURE

A. INTRODUCTION

By identifying the top-level architecture tradespace and then using the methodology from the First International Assessment of Space Solar Power presented in Chapter II, the requirements of the system will be identified, a functional architecture created, the interrelationship of the elements in the functional architecture modeled, and figures of merit identified for each element. This methodology will allow a quantitative analysis of different architectures and facilitate varying the parameters to understand the sensitivity of the architecture to changing parameters or to seek system cost reductions.

B. ARCHITECTURE TRADESPACE

As identified throughout the background information, there are many system level trades that must be conducted. These are represented in Figure 20. The first system trade is the location of the lunar outpost at either the South Pole or South Pole-Aitkin Basin. The choice at this tradespace is to maximize scientific value but as identified later in Chapter IV, results in a larger constellation size to ensure 100% coverage if the solar power satellite concept for power generation is used. Given the outpost location, the next tradespace is the choice of power generation technology. The options here include surface photovoltaics with complementary batteries, nuclear power, or a solar power satellite. The tradespace includes the mass of the batteries in the surface photovoltaics and the safety issues with the nuclear power as discussed in the background chapter.

Given the selection of solar power satellite, the type of orbit is the next architecture trade. The three orbits identified, low lunar orbit, halo orbit, and frozen elliptical orbit will have several impacts on the architecture. If the low lunar orbit is selected, a large number of satellites would be necessary to ensure coverage since each satellite would have only limited access per orbit to the lunar outpost; however, the size of the solar power satellite transmitter and lunar receiver would be at a minimum. A halo orbit at the L1 Lagrange point at a distance of 57,000 km would drastically increase the solar power satellite mass through the size of the required transmitter and lunar receiver.

As identified in the background chapter, the frozen elliptical orbit provides the best coverage and orbital distance.

With the decision of a solar power satellite the coverage requirement impacts the architecture. As discussed later in Chapter IV, if the requirement is for continuous power with 100% coverage at the lunar outpost, then the number of satellites in the constellation will increase. As an option in the architecture but not analyzed in this thesis, is for a smaller constellation size augmented by a surface energy storage system. The next issue in the tradespace is the type of wireless power transmission of either microwaves or laser. The choice of wireless power transmission will have an impact on the mass of the satellite and the size of the lunar receiver which both increase system cost. Finally, with the choice of wireless power transfer influencing the mass of the solar power satellite, the tradespace for the launch vehicle can be understood.

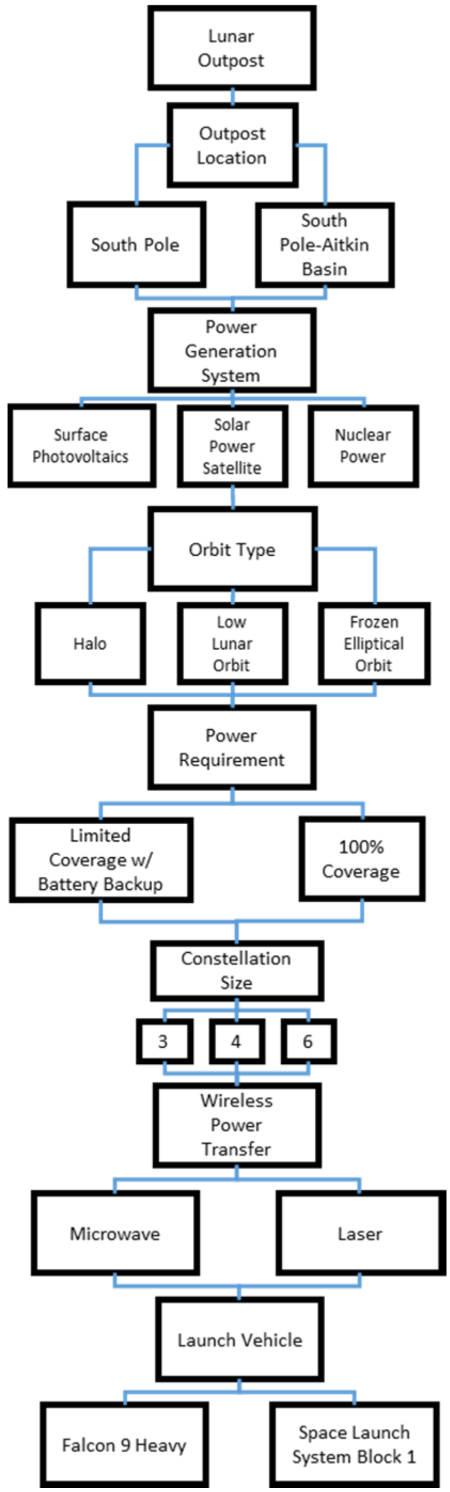


Figure 20. Architecture Trades

C. REQUIREMENTS

The operational view identifies the unique elements of the scenario and the interaction between the architecture and the environment. The operational view in Figure 21 is a depiction of the scenario of a solar power satellite in orbit around the moon collecting and converting sunlight and transmitting this energy via wireless power transfer to a lunar outpost receiver. The user of this system will be astronauts operating from a lunar base. The requirements for electrical power at the lunar base is for a continuous energy supply with minimum energy storage requirements.

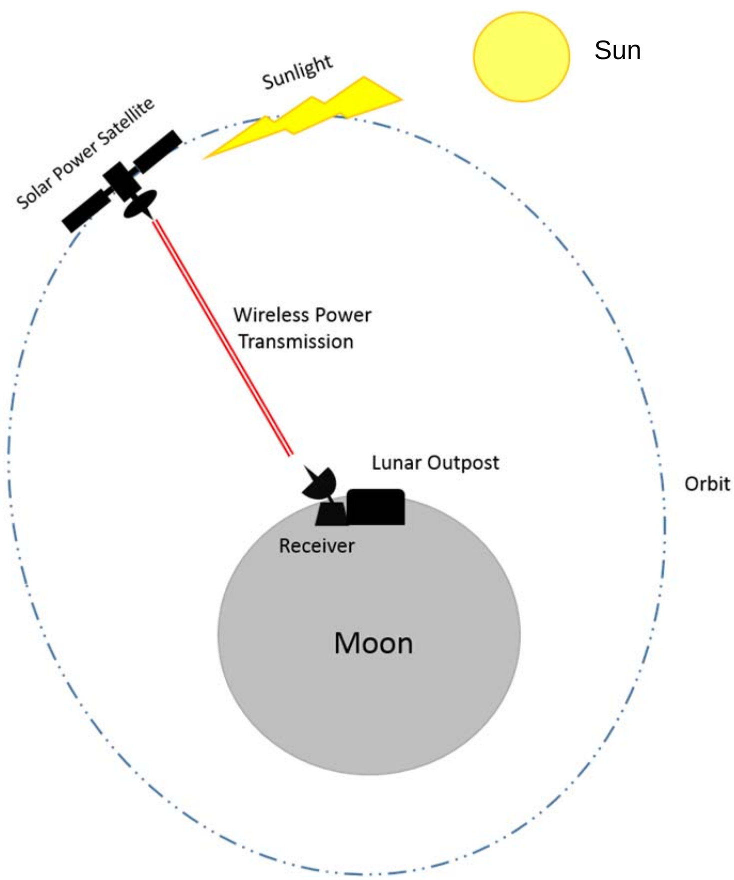


Figure 21. Operational View

The interaction between the environmental inputs and system elements can be visualized with the use case diagram in Figure 22. Ultimately, from the initial system

input of sunlight, the final system output is usable electricity to astronauts. In between the initial input and final output, the solar power satellite will have to collect, covert, target, and transmit energy to a lunar receiver which has to collect and convert the energy.

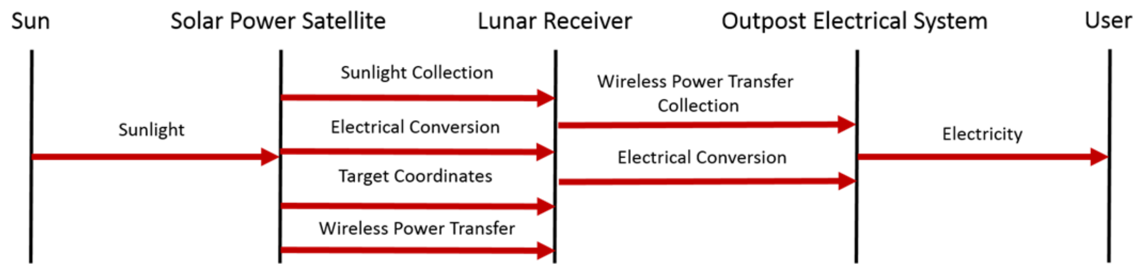


Figure 22. System Use Case

D. FUNCTIONAL ARCHITECTURE

A modified version of the functional architecture will be used from the First International Assessment of Space Solar Power illustrated in Figure 23. The functional architecture simplifies the solar power satellite into three elements, the solar power generation system, the SPS platform, and the wireless power transmission system. The SPS platform technologies system includes the power management and distribution system, the satellite supporting structures, the gimbaling system, and thermal management system. Outside of the solar power satellite are the launch vehicle and lunar receiver.

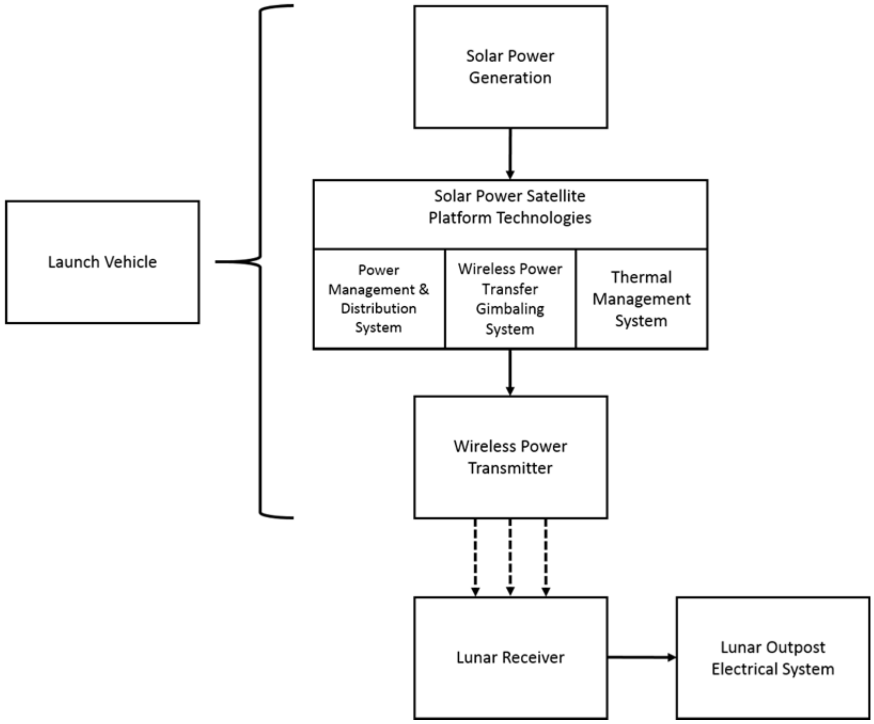


Figure 23. SPS Functional Architecture

E. INTERRELATIONSHIP OF FUNCTIONAL ARCHITECTURE

Understanding the relationship between the elements in the functional architecture allows the comparison of different concepts required to understand the available tradespace. The relationship between the elements in the functional architecture can be seen in Figure 24.

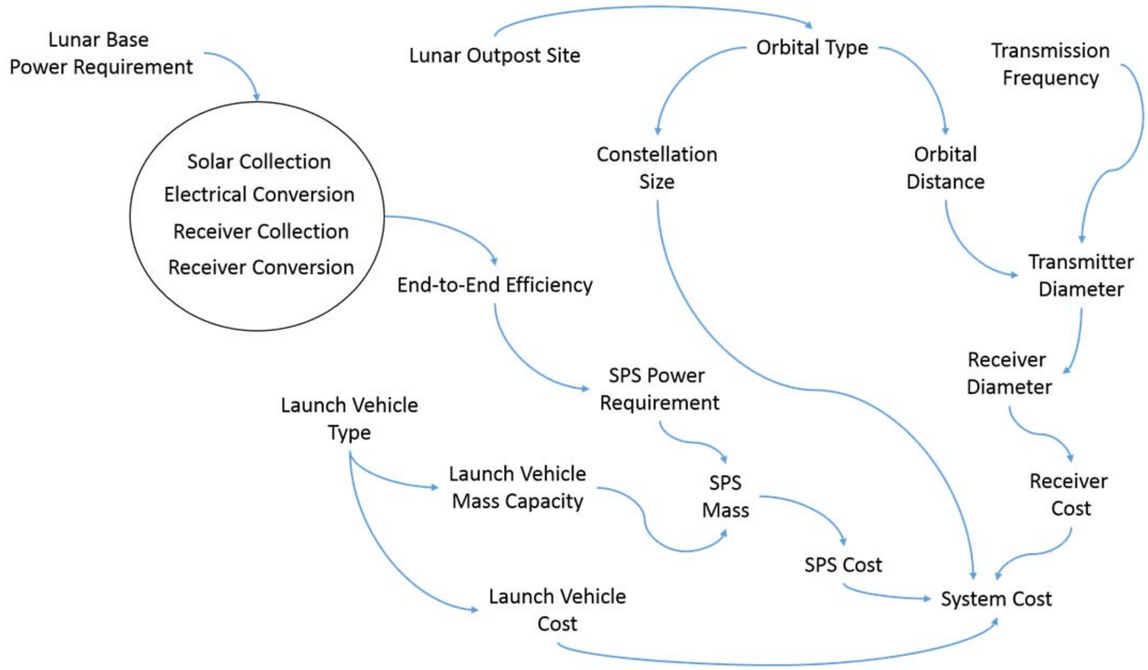


Figure 24. Relationship of Functional Elements

From Figure 24 the choice of lunar outpost site and the power requirement of the outpost are the initial parameters identified. Then, given the inefficiencies in the solar power generation, wireless power transmission, and lunar receiver, an end-to-end efficiency can be calculated. This inefficiency will burden the solar power satellite increasing the amount of power the satellite will have to produce to deliver the required power level at the outpost. From the SPS power requirement, the mass of the satellite can be determined which is bounded by the trans-lunar injection mass capacity of the launch vehicle. The cost of the SPS system can be calculated when the mass of the satellite is determined and the cost of the satellite, lunar receiver, and launch vehicle will inform the total system cost.

Given the lunar outpost site and the orbit type, the constellation size can be determined to provide 100% coverage of the outpost. With the orbit type selected, the distance from the lunar outpost is calculated, and with the selection of the transmission frequency, the size of the transmitter can be determined. Since the transmitter and receiver are related through equation (3), the size of the receiver diameter can be

calculated. The receiver diameter is related to the receiver cost which influences the total system cost. System cost is used to determine the choice of satellite architecture.

F. ANALYSIS PLAN - FIGURES OF MERIT

The following figures of merit for elements of the solar power satellite system will be used to analyze the feasibility of the microwave and laser concepts and the architecture trades. The lunar outpost location is the South Pole-Aitkin Basin at 56˚ South, 180˚ East (Kring 2012). Given this location and the use of the frozen elliptical orbit (Equation 1) the worst case distance from the solar power satellite to the lunar outpost is 9881 kilometers. Two figures of merit for the solar power satellite are outlined in Table 2.

Table 2. Solar Power Satellite Figures of Merit

Figure of Merit	Value	Reference
SPS Power per Unit Mass	0.2 kW/kg	(Mankins 2011)
SPS Cost per Unit Mass	100,000 $/kg	(Mankins 2011)

Table 3 shows the figures of merit for the microwave solar power satellite with the appropriate microwave receiver. The first four microwave figures of merit give the end-to-end efficiency of the microwave concept. The remaining three figures of merit will show the impact the microwave receiver has on the feasibility of the microwave SPS concept.

Table 3. Microwave Figures of Merit

Figure of Merit	Value	Reference
RF to DC Conversion Efficiency	72%	(Marzwell 2008)
RF Collection Efficiency	93%	(Marzwell 2008)
DC to RF Conversion Efficiency	78%	(Tanwar 2013)
Solar Collection to DC Efficiency	25%	(O'Neill 2006)
Receiver Conversion Density	1 W/cm^2	(Marzwell 2008)
Receiver Mass per Unit Area	0.16 kg/m^2	(Brown 1987)
Receiver Lunar Cost per Unit Mass	100,000 $/kg	(Eckart 1996)

Table 4 shows the laser figures of merit for the laser solar power satellite with appropriate laser receiver. The first four laser figures of merit give the end-to-end efficiency of the laser concept. The remaining three figures of merit will show the impact of the laser receiver on the feasibility of the laser SPS concept.

Table 4. Laser Figures of Merit

Figure of Merit	Value	Reference
Laser to DC Conversion Efficiency	45%	(O'Neill 2006)
Laser Collection Efficiency	92%	(O'Neill 2006)
DC to Laser Conversion Efficiency	80%	(Summerer 2009)
Solar Collection to DC Efficiency	25%	(O'Neill 2006)
Receiver Conversion Density	0.069 W/cm^2	(O'Neill 2006)
Receiver Mass per Unit Area	0.86 kg/m^2	(O'Neill 2006)
Receiver Lunar Cost per Unit Mass	100,000 $/kg	(Eckart 1996)

Table 5 shows the figures of merit for the Falcon 9 Heavy launch vehicle. The number of launches will be equal to the number of satellites and the available launch vehicle mass from trans-lunar orbit to low lunar orbit will be calculated when using the hall current thrusters (HCT).

Table 5. Launch Vehicle Figures of Merit

Figure of Merit	**Value**	**Reference**
Launch Vehicle Cost	\$158M per Launch	(Skran 2015)
Launch Vehicle Mass to TLI	13,200 kg	(Skran 2015)
Delta-V Required from TLI to LLO	800 m/s	(Biesbroek 2000)
HCT Specific Impulse	2,100 sec	(Welander 2001)

IV. DATA ANALYSIS

A. INTRODUCTION

In this chapter the background information on lunar orbits, the theoretical physics identified for microwave and laser wireless power transfer, and the key figures of merit for selected technologies will be used to analyze different solar power satellite concepts. The first analysis presented will be the orbital coverage and access data generated by modeling the outpost location and frozen elliptical orbit in STK. The second analysis presented will be the system cost, mass and power thread calculations using a SPA Basin lunar outpost with a four satellite constellation employing a microwave or laser solar power satellite. The final analysis will show the sensitivity of the SPS concepts to changes in key parameters that drive down cost and mass. This analysis will be used to draw conclusions in Chapter V.

B. ANALYSIS OF ARCHITECTURES

The orbital analysis was conducted for two lunar outpost locations, the South Pole and the South Pole–Aitken Basin (SPA Basin) (56˚ South, 180˚ East). The South Pole–Aitken basin is a huge impact crater on the far side of the Moon roughly 2,500 kilometers (1,600 mi) in diameter and 13 kilometers (8.1 mi) deep. It is one of the largest known impact craters in the solar system and as identified in the background information a science-rich environment for lunar exploration.

The first modeling conducted was to determine the lighting conditions at the South Pole and SPA Basin to determine if the use of surface solar photovoltaics was an architecture option. Figure 25 and Figure 26 from STK show the lighting conditions at the South Pole and SPA Basin only receiving sunlight 43% of the time, each with different lighting intervals. The use of surface photovoltaics with an energy storage system at these locations would require a large mass energy storage system as mentioned in the background chapter.

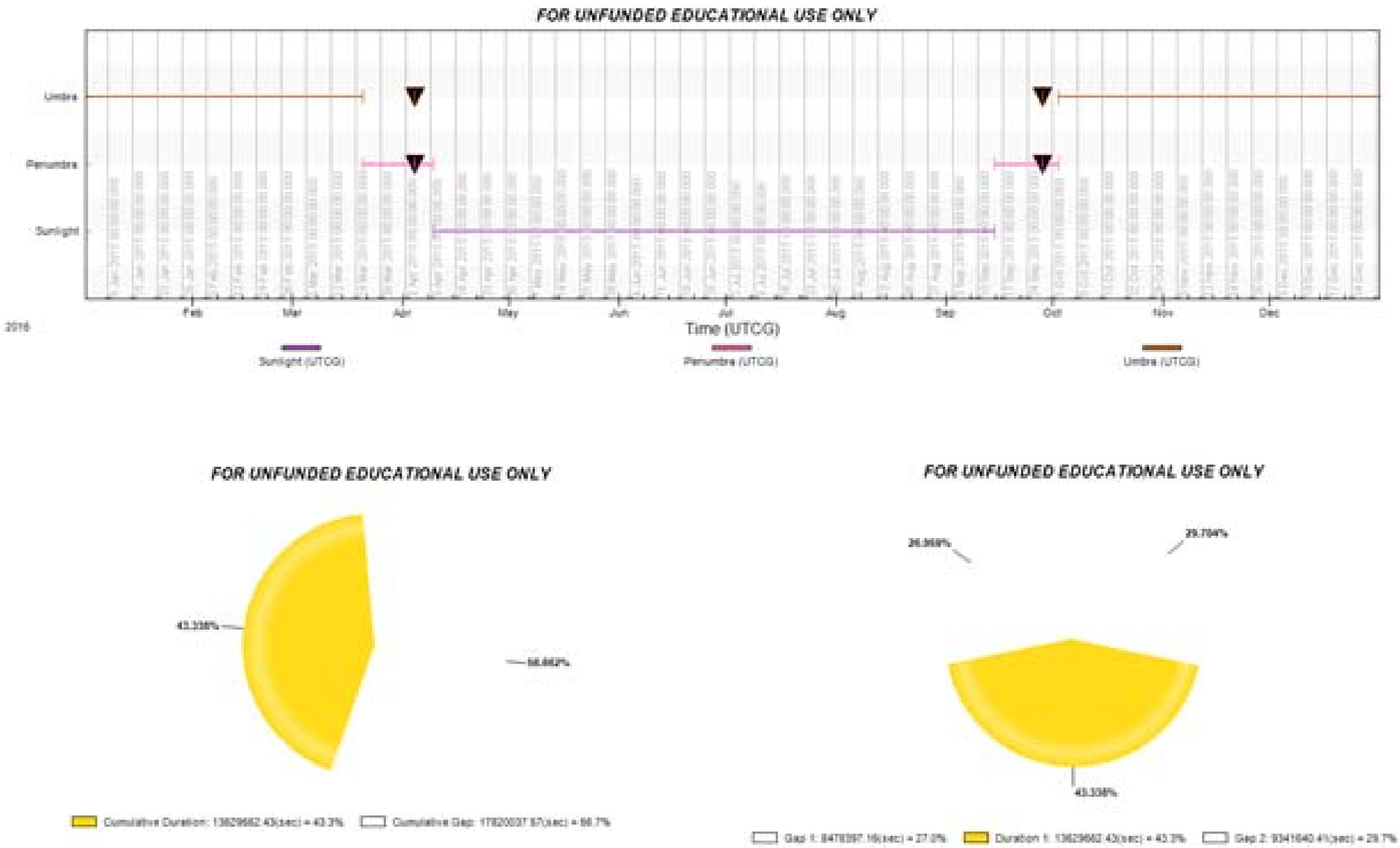


Figure 25. South Pole Lighting Times

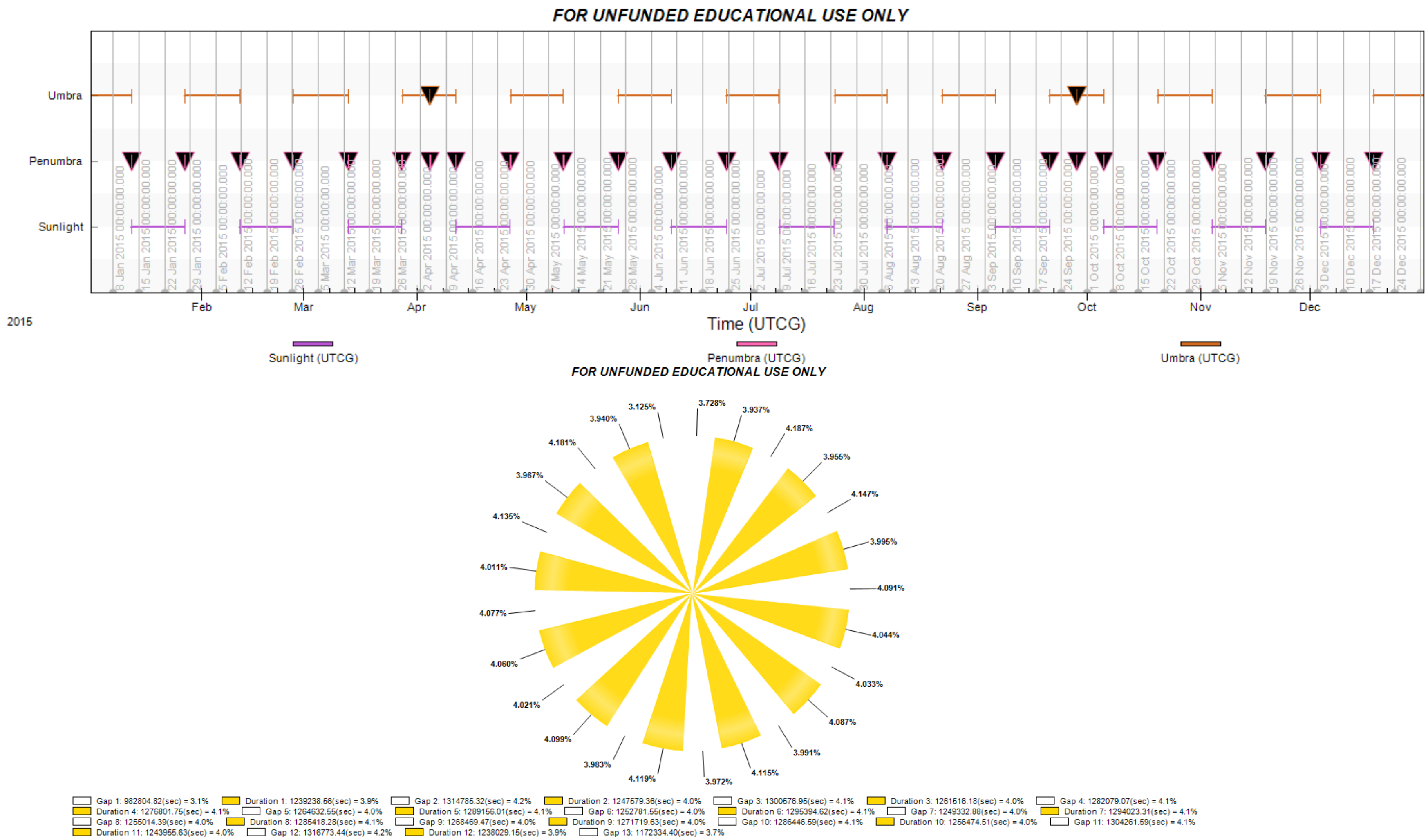


Figure 26. SPA Basin Lighting Times

The orbit type modeled was the frozen elliptical orbit as defined in Equation (1). This orbit has several unique features including a 10-year life with few perturbations from the three-body orbital dynamics of the Earth-Moon-Sun system. This resulted in a three-satellite constellation which provided continuous coverage for a South Pole lunar outpost with a mean pass of 10.6 hours per satellite and often continuous two-fold satellite coverage. When the minimum elevation angle for the lunar outpost is changed to 20˚ for access to the satellite the mean pass is 9.0 hours per satellite. See Table 6. The 20˚ minimum elevation constraint was selected because the lunar outpost will most likely be at the bottom of crater.

Table 6. South Pole Satellite Access

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Facility-SouthPole-To-Satellite-SouthPoleSat1: Access Summary Report

SouthPole-To-SouthPoleSat1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (hr)
Global Statistics				
Min Duration	438	29 Aug 2015 17:59:37.827	29 Aug 2015 19:53:05.692	1.891
Max Duration	660	29 Dec 2015 10:26:10.139	29 Dec 2015 19:30:26.395	9.071
Mean Duration				9.030
Total Duration				5995.906

The 20˚ minimum elevation angle was chosen as a more realistic constraint on the lunar outpost and the three-satellite frozen elliptical orbit constellation still maintains continuous coverage as seen in Figure 27. The constellation as modeled is shown in Figure 28 and the associated ground track in Figure 29. The worst case range from the satellite using this orbit to the South Pole lunar outpost is 9,108 km as seen in Table 7.

Table 7. South Pole Satellite Azimuth, Elevation, and Range

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Facility-SouthPole-To-Satellite-SouthPoleSat1: Inview Azimuth, Elevation, & Range

SouthPole-To-SouthPoleSat1 - AER reported in the object's default AER frame

	Time (UTCG)	Azimuth (deg)	Elevation (deg)	Range (km)
Global Statistics				
Min Elevation	20 Feb 2015 03:49:40.696	257.470	15.000	5884.851082
Max Elevation	29 Dec 2015 15:01:17.046	34.329	48.729	9096.465922
Mean Elevation			26.190	
Min Range	31 Dec 2015 11:04:11.754	76.558	15.000	5860.701353
Max Range	11 Jan 2015 09:46:18.295	354.121	48.108	9108.400632
Mean Range				6986.897932

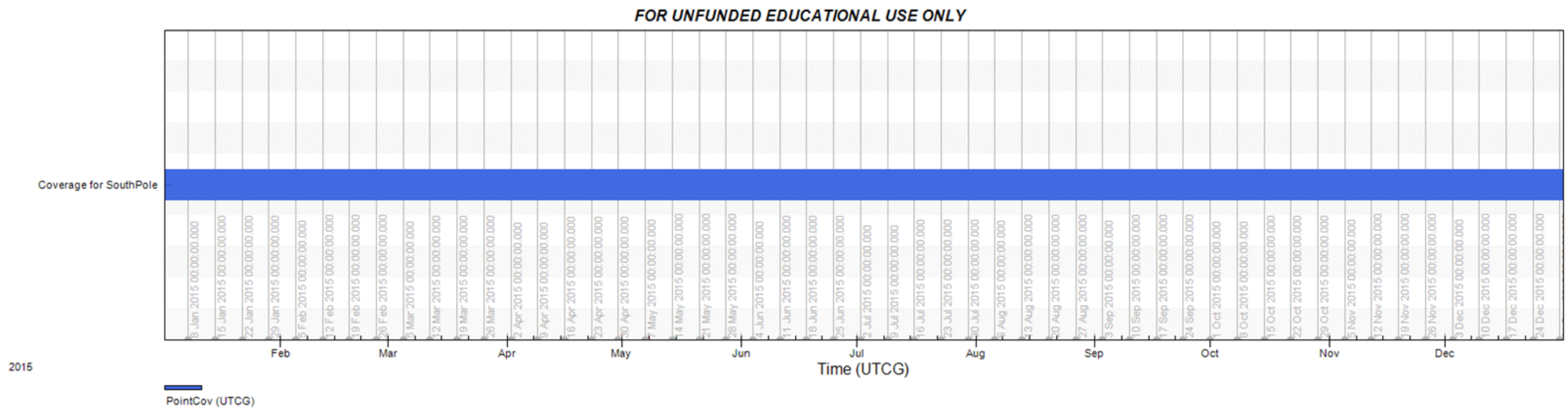

Figure 27: Coverage for South Pole using Three-satellites

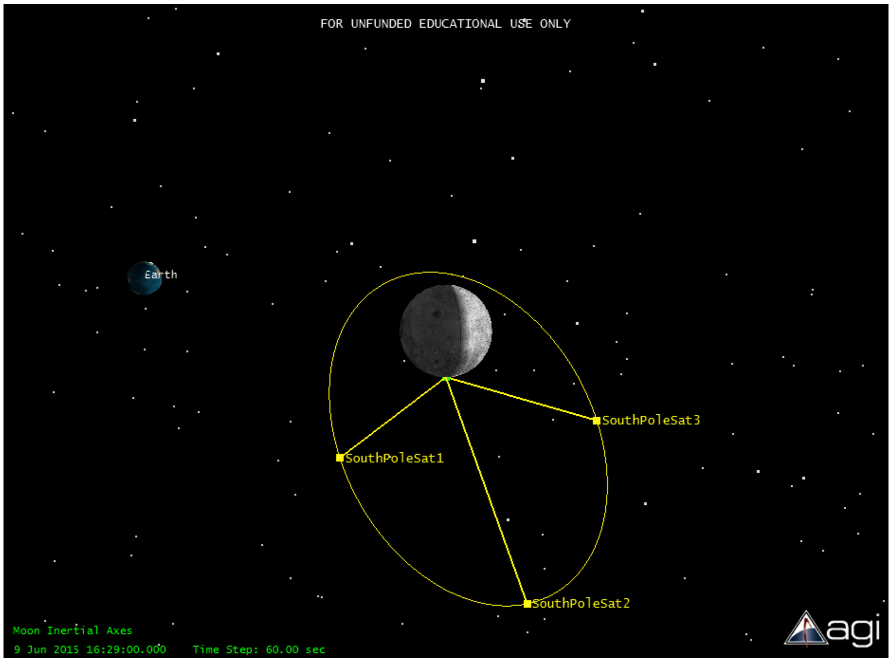

Figure 28: South Pole Three-Satellite Constellation

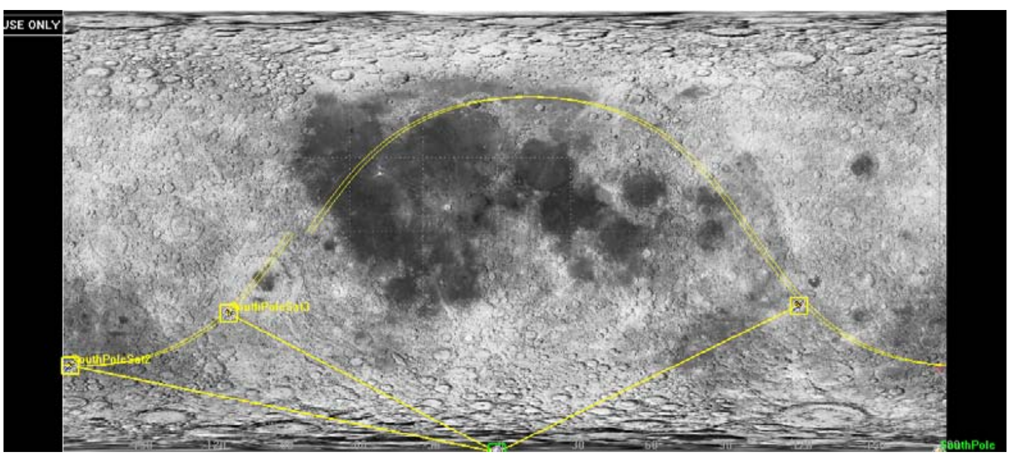

Figure 29: Ground Track for Three-Satellite Constellation South Pole Coverage

A challenge arises, however, when this orbit is used for the South Pole-Aitken Basin lunar outpost with a 20˚ minimum elevation angle. The SPA lunar outpost is located at 56˚ S, 180˚ E, and therefore rotates out of view of the classic three-satellite constellation used for the South Pole lunar outpost. Table 8 and Figure 30 show an approximate 90% coverage where the classic three-satellite constellation drops coverage

for a few days each month. Figure 31 shows the ground track when the classic three-satellite constellation does not cover the SPA lunar outpost.

Table 8. Coverage Statistics for SPA Basin Outpost using Class Three-Satellite Constellation

```
                                                                              09 Aug 2015 15:07:29
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Satisfaction Intervals

            FOM Properties
-------------------------------------
Simple Coverage FOM
Satisfaction is not enabled
FOM value range check is not enabled.


Satisfaction Intervals for SPAoutpost
-------------------------------------

                    Interval Start (UTCG)        Interval End (UTCG)       Duration (hr)    Percent
Global Statistics
-----------------
Min Duration         6 May 2015 02:58:30.413     6 May 2015 03:06:50.969            0.14       0.00
Max Duration        16 Nov 2015 09:55:17.012     9 Dec 2015 19:53:11.123          561.97       6.43
Total Duration                                                                   7878.27
Total Percent                                                                                 90.18
```

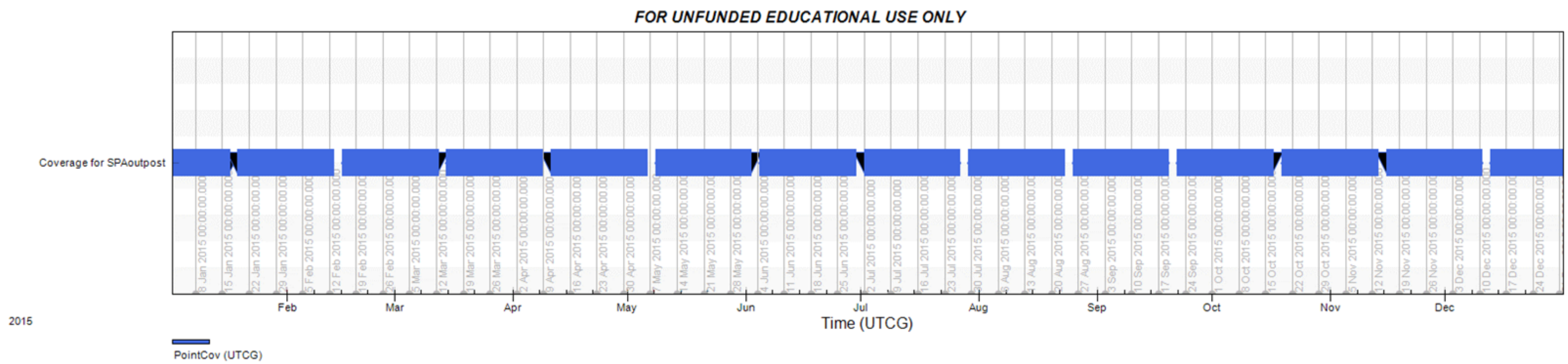

Figure 30: South Pole-Aitken Basin Coverage using Three-Satellites

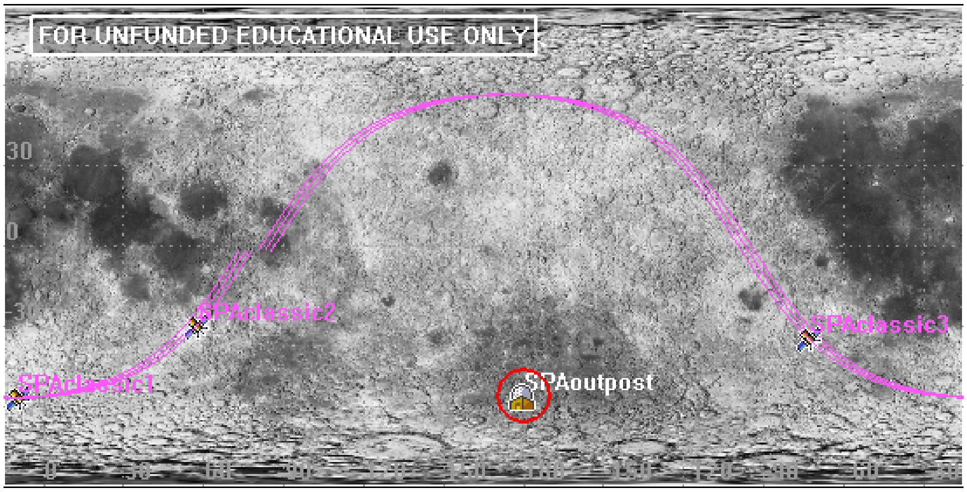

Figure 31: Ground Track for Three-Satellite Constellation for SPA Coverage

In order to cover the SPA Basin lunar outpost, it is required to add another plane of satellites to the constellation. Initially, two planes of three satellites each was designed and analyzed. Figure 32 show continuous coverage of the SPA lunar outpost when using this constellation design. Figure 33 shows the six satellite architecture with two planes of satellites offset by 180˚ and Figure 34 shows the resulting ground track.

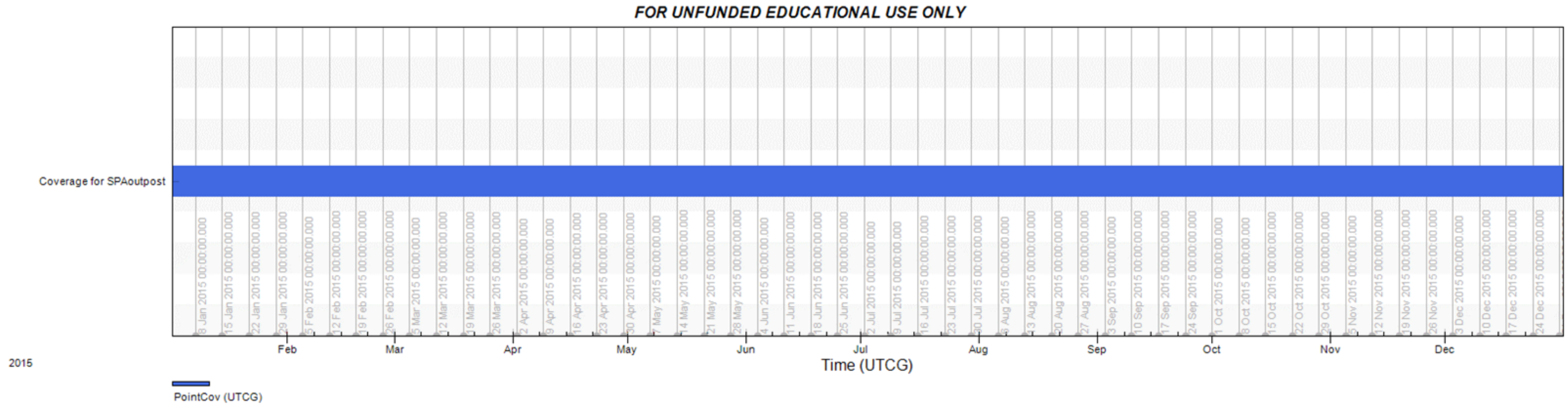

Figure 32: SPA Coverage Using Six-Satellites (Three in each plane offset by 180˚)

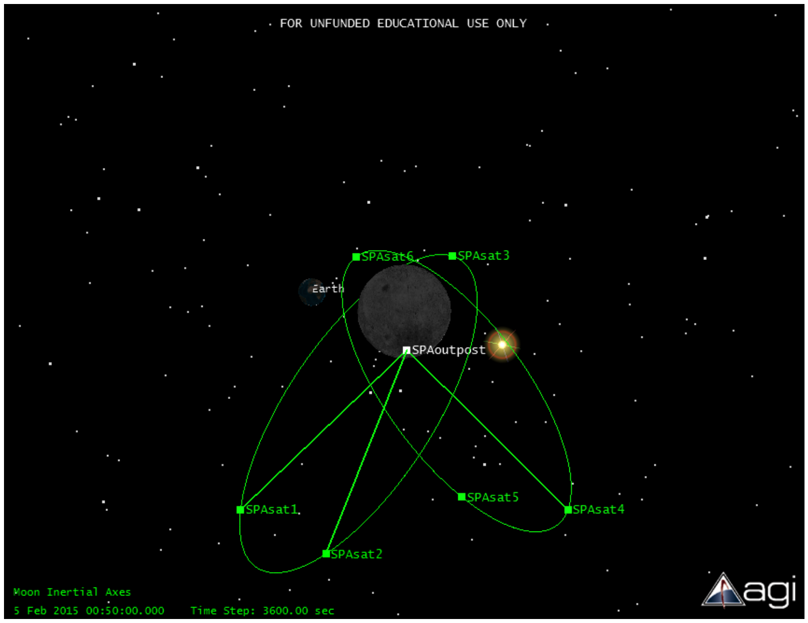

Figure 33: SPA Six-Satellite Constellation

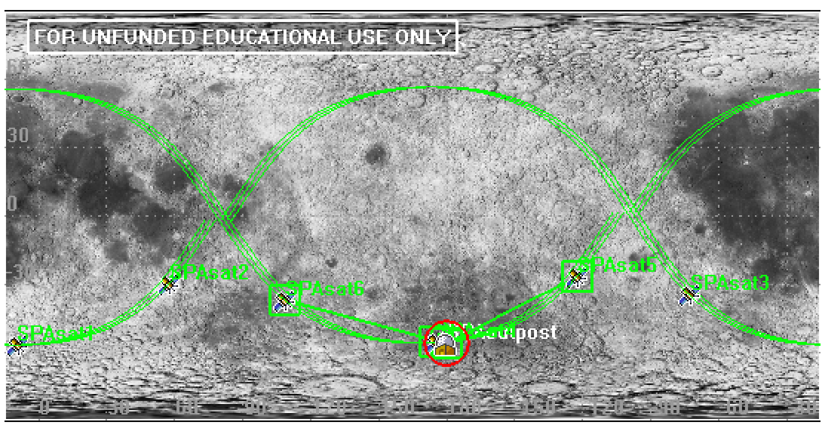

Figure 34: Ground Track for Six-Satellite Constellation for SPA Coverage

Even though this constellation of two planes with three satellites each provides continuous coverage, it leads to a large system cost. Therefore, another satellite constellation was designed and analyzed with two planes of two satellites to determine if it provided continuous coverage with reduced constellation size. Figure 35 shows continuous coverage of the SPA lunar outpost given the two-plane two-satellite design. Figure 36 shows the two-plane two-satellite design and Figure 37 shows the resulting ground track.

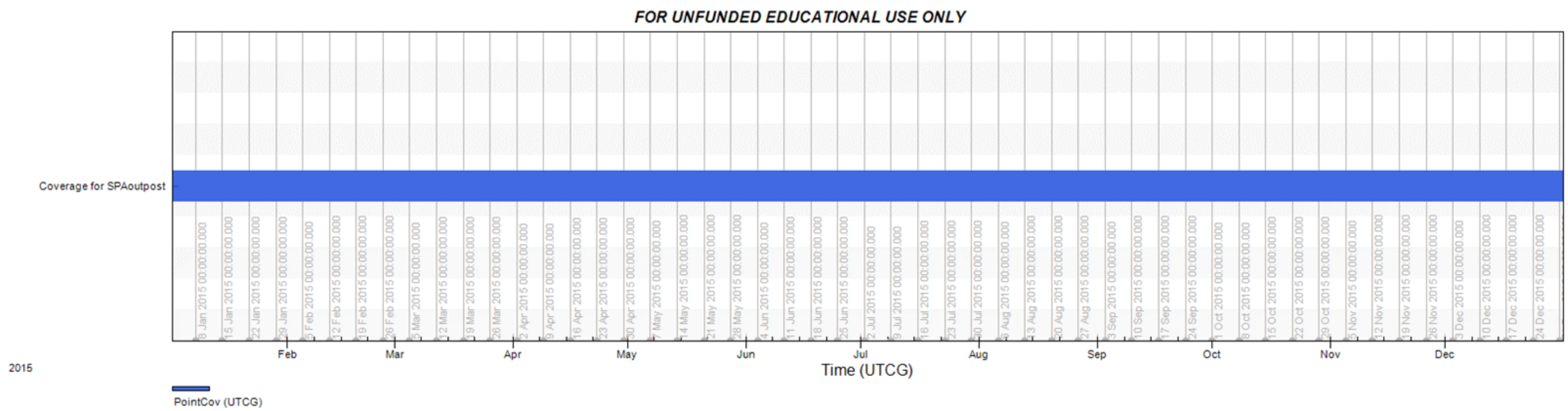

Figure 35: SPA Coverage Using Four Satellites (Two in each plane offset by 180˚)

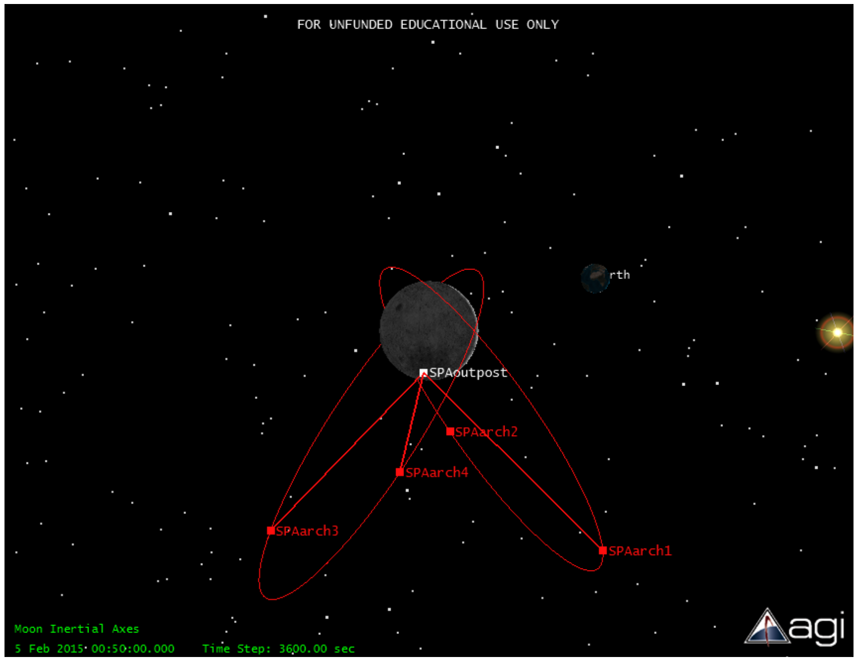

Figure 36: SPA Four-Satellite Constellation

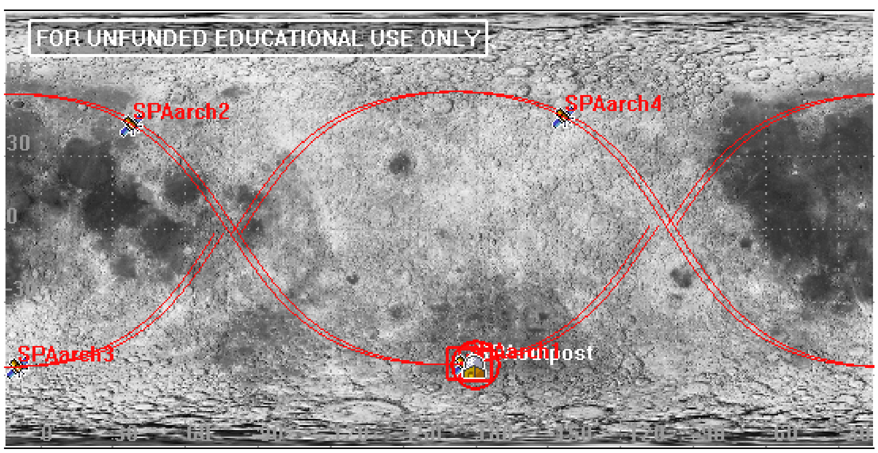

Figure 37: Ground Track for Four-Satellite Constellation for SPA Coverage

Using the two-plane two-satellite constellation, the mean pass for each satellite is 7.7 hours as seen in Table 9, and the worst case range from the satellite to the lunar outpost is 9881 km is shown in Table 10.

Table 9. SPA Basin Satellite Access

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Facility-SPAoutpost-To-Satellite-SPAarch1: Access Summary Report

SPAoutpost-To-SPAarch1

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (hr)
Global Statistics				
Min Duration	180	22 Apr 2015 00:22:25.262	22 Apr 2015 00:52:44.855	0.505
Max Duration	25	17 Jan 2015 19:10:07.114	18 Jan 2015 04:52:07.674	9.700
Mean Duration				7.658
Total Duration				4503.145

Table 10. SPA Basin Satellite Azimuth, Elevation, and Range

FOR UNFUNDED EDUCATIONAL USE ONLY
Facility-SPAoutpost-To-Satellite-SPAarch1: Inview Azimuth, Elevation, & Range

SPAoutpost-To-SPAarch1 - AER reported in the object's default AER frame

	Time (UTCG)	Azimuth (deg)	Elevation (deg)	Range (km)
Global Statistics				
Min Elevation	20 Jan 2015 10:01:49.383	58.253	15.000	4383.096419
Max Elevation	13 Mar 2015 09:20:04.182	0.096	89.624	8717.585355
Mean Elevation			27.123	
Min Range	11 Jan 2015 04:10:38.421	359.486	15.000	3517.115381
Max Range	30 Nov 2015 12:01:51.560	161.911	15.000	9881.065731
Mean Range				7174.362137

After modeling three different satellite architectures for the South Pole and SPA Basin, the resulting satellite constellation which provides coverage for the highest subjective value of scientific research at minimal additional cost is the four-satellite constellation with two planes of two satellites at the SPA Basin. This satellite architecture was used in the system mass, cost, and power thread analysis.

1. System Mass, Cost, and Power Thread Analysis

With the satellite architecture and lunar outpost location chosen, the overall system mass, cost, and power thread analysis can be completed for the wireless power transmission options of microwave or laser. The inputs to this analysis included frequency/wavelength, worst case satellite range from the orbital analysis, the theoretical physics from the background chapter, and figures of merit from the system architecture

section. The analysis starts by using the lunar outpost power requirement and works backwards through the power thread chain to calculate an end-to-end efficiency. Using the end-to-end efficiency and SPS power requirement, the mass and cost of a single SPS is determined. The next parameter calculated is the receiver and transmitter sizes, and the cost of the lunar receiver considered. Finally, the launch cost and total system cost are calculated. The final output of the analysis is the total system cost which is used to determine the feasibility of the microwave and laser solar power satellite concepts. Table 11 and Table 12 show the analysis for the microwave and laser calculations, respectively.

Table 11. Microwave SPS System Analysis

Microwave System Analysis		
Constraits	Value	Units
Frequency	94	GHz
Wavelength	0.0032	m
Spacecraft Range	9881	km

Step 1: Calculate the end-to-end power efficiency and SPS power requirement				
		Value	Units	
Lunar Outpost Power Requirement		100	kW	A
RF to DC Conversion Efficiency	Figure of Merit	72%	%	B
Power Required due to RF to DC Conversion Efficiency	C=A/B	139	kW	C
RF Collection Efficiency	Figure of Merit	93%	%	D
Power Required due to RF Collection Collection Efficiency	E=C/D	149	kW	E
Amount of Power in the Main Lobe of the Beam	Figure of Merit	84%	%	F
Power Required due to Power in the Main Lobe of the Beam	G=E/F	178	kW	G
DC to RF Conversion Efficiency	Figure of Merit	78%	%	H
Power Required due to DC to RF Conversion Efficiency	I=G/H	228	kW	I
Solar to DC Conversion Efficiency	Figure of Merit	25%	%	J
Power Required due to Solar to DC Conversion Efficiency	K=I/J	912	kW	K
End-to-End Power Efficiency	L=B*D*F*H*J	11.0%	%	L
SPS Power Requirement	M=K	912	kW	M
Step 1 Results: The end-to-end efficiency for the microwave power thread is 11% which results in a SPS power requirement of 912 kilowatts to deliver 100 kilowatts to the lunar outpost.				

Step 2: Calculate the mass and cost of the SPS.				
		Value	Units	
SPS Power per Unit Mass	Figure of Merit	0.2	kW/kg	N
SPS Mass	O=M/N	4.6	mT	O
SPS Cost per Unit Mass	Figure of Merit	100,000	$/kg	P
SPS Satellite Cost	Q=O*P	456	$M	Q

Step 3: Calculate the optimum size of the microwave receiver

	Value	Units	
Conversion Density	1	Watt/cm^2	R
1 m^2	10,000	cm^2	S
E in Watts	149343	Watts	T
U=T/(R*S)	15	m^2	U

Step 3 Result: The optimum microwave receiver area is 15 m^2

Step 4: Calculate the size of the SPS transmitter given the optimum size of the microwave receiver

			Value	Units	
	Receiver Diameter	V=SQRT(U*4/π)	4.4	m	V
		Spacecraft Apogee	9881	km	W
		Frequency	94	GHz	X
		Speed of Light	3.00E+08	m/s	Y
	Wavelength	Z=Y/X	0.0032	m	Z
Using Main Lobe Diffraction Equation	Transmitter Diameter	2.44=(V*AA)/(Z*W)	17646	m	AA

Step 4 Results: Using the optimum size of the lunar receiver results in an unacceptable large transmitter diamter of 17,646 meters. Choosing to balance the size of the transmitter and receiver diameters which at 94 GHz and a distance of 9881 kilometers results in a transmitter diameter size of 277 meters and a receiver diameter size of 278 meters.

Step 4.1: Balance the size of the lunar receiver and SPS transmitter diameters

			Value	Units	
	Transmitter Diameter		277	m	BB
Using Main Lobe Diffraction Equation	Receiver Diamter	2.44=(CC*BB)/(Z*W)	278	m	CC

Step 4.1: Even when balancing the diameters of the transmitter and receiver, a SPS transmitter diameter of 277 is exceptionally large. A more reasonable size of the transmitter based on the largest commerical antenna launched of 22 meters on the SkyTerra-1 satellite is selected.

Step 4.2: Using the largest antenna of 22 meters launched on the SkyTerra-1 satellite

			Value	Units	
	Transmitter Diameter		22	m	DD
Using Main Lobe Diffraction Equation	Receiver Diamter	2.44=(P*R)/(N*K)	3498	m	EE

Step 4.2 Results: The 22 meter transmitter diameter results in a lunar receiver diameter of 3500 meters. For the purposes of these calculations the 278 meter reciever diameter will be used as the 3500 meter diameter results in a significant receiver mass and cost.

Step 5: Calculate the mass and cost of the lunar receiver using the 278 meter diameter

		Value	Units	
Receiver Area	FF=π*(CC/2)^2	60604	m^2	FF
Receiver Mass per Unit Area	Figure of Merit	0.16	kg/m^2	GG
Receiver Mass	HH=FF*GG	9697	kg	HH
Receiver Lunar Cost per Unit Mass	Figure of Merit	100,000	$/kg	II
Receiver Lunar Cost	JJ=HH*II	970	$M	JJ

Step 5 Results: Given the balance between the size of the SPS transmitter and lunar receiver, the cost of the receiver is $970 million which is more than the cost for a single SPS satellite

Step 6: Calculate Trans-Lunar Orbit (TLI) to Low Lunar Orbit (LLO)

		Value	Units	
Mass Initial	Figure of Merit	13200	kg	KK
Delta-V	Figure of Merit	800	m/s	LL
Specific Impulse	Figure of Merit	2100	s	MM
Earth Gravitational Constant	Constant	9.807	m/s^2	NN
Mass Final	OO=KK/e^(LL/MM*NN)	12697	kg	OO
Change in Mass	PP=KK-OO	503	kg	PP

Step 6 Results: Using the TLI mass of the Falcon 9 Heavy of 13.2 metric tons and solving for Mass final results in an available payload mass of 12,697 kg.

Step 7: Calculate Launch Cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	QQ
Number of Launches	RR=QQ	4	Launches	RR
Cost per Launch	Figure of Merit	158	$M/Launch	SS
Launch Cost	TT=RR*SS	632	$M	TT

Step 7 Results: The total launch cost for 4 satellites is $632 million dollars. The Falcon 9 Heavy cost was used and even though the mass of the Falcon 9 Heavy is 13,200 kg which could allow the launch vehicle to carry 2 satellites, this thesis assumes that the volume of the SPS especially given a 277 meter diameter transmitter, would use all of the Falcon 9 Heavy's volume.

Step 8: Calculate the total system cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	UU
Cost per Satellite	VV=Q	456	$M	VV
Constellation Cost	WW=UU*VV	1823	$M	WW
System Cost	XX=JJ+TT+QQ	3425	$M	XX

Step 8 Results: Using the cost of the lunar receiver, the launch cost, and the constellation cost, the total system cost for the microwave solar power satellite concept is $3.4 billion dollars.

Table 12. Laser SPS System Analysis

Laser System Analysis		
Constraits	Value	Units
Wavelength	805	m
Spacecraft Range	9881	km

Step 1: Calculate the end-to-end power efficiency and SPS power requirement

		Value	Units	
Lunar Outpost Power Requirement		100	kW	A
Laser to DC Conversion Efficiency	Figure of Merit	45%	%	B
Power Required due to Laser to DC Conversion Efficiency	C=A/B	222	kW	C
Laser Collection Efficiency	Figure of Merit	92%	%	D
Power Required due to Laser Collection Efficiency	E=C/D	242	kW	E
DC to Laser Conversion Efficiency	Figure of Merit	80%	%	F
Power Required due to DC to Laser Conversion Efficiency	G=E/F	302	kW	G
Solar to DC Conversion Efficiency	Figure of Merit	25%	%	H
Power Required due to Solar to DC Conversion Efficiency	I=G/H	1208	kW	I
End-to-End Power Efficiency	J=B*D*F*H	8.3%	%	J
SPS Power Requirement	K=I	1208	kW	K

Step 1 Results: The end-to-end efficiency for the laser power thread is 8.3% which results in a SPS power requirement of 1208 kilowatts to deliver 100 kilowatts to the lunar outpost.

Step 2: Calculate the mass and cost of the SPS.

		Value	Units	
SPS Power per Unit Mass	Figure of Merit	0.2	kW/kg	L
SPS Mass	M=K/L	6.0	mT	M
SPS Cost per Unit Mass	Figure of Merit	100,000	$/kg	N
SPS Satellite Cost	O=M*N	604	$M	O

Step 3: Calculate the size of the laser receiver

	Value	Units	
Conversion Density	0.069	Watt/cm^2	P
1 m^2	10,000	cm^2	Q
E in Watts	241546	Watts	R
S=R/(P*Q)	350	m^2	S

Step 3 Result: The laser receiver is 350 m^2

Step 4: Calculate the size of the SPS transmitter radius given the size of the laser receiver

			Value	Units	
	Beam Waist Radius		1.0	m	V
	Wavelength		805	nm	W
	Spacecraft Apogee		9881	km	X
Transmitter Radius	Y=V*SQRT(1+(W*X/(PI()*V^2))^2)		2.72	m	Y
	Transmitter Area	Z=PI()*r^2	23	m^2	Z

Step 4 Results: Given the laser receiver area and the radius of the laser beam waist, the transmitter radius is 2.72 meters with a total area of 23 m^2

Step 5: Calculate the mass and cost of the lunar receiver

		Value	Units	
Receiver Area	AA=S	350	m^2	AA
Receiver Mass per Unit Area	Figure of Merit	0.86	kg/m^2	BB
Receiver Mass	CC=AA*BB	301	kg	CC
Receiver Lunar Cost per Unit Mass	Figure of Merit	100,000	$/kg	DD
Receiver Lunar Cost	EE=CC*DD	30	$M	EE

Step 5 Results: Both the lunar receive and transmitter diameters are reasonable which leads to a receiver cost of $30 million dollars

Step 6: Calculate Trans-Lunar Orbit (TLI) to Low Lunar Orbit (LLO)

		Value	Units	
Mass Initial	Figure of Merit	13200	kg	FF
Delta-V	Figure of Merit	800	m/s	GG
Specific Impulse	Figure of Merit	2100	s	HH
Earth Gravitational Constant	Constant	9.807	m/s^2	II
Mass Final	JJ=FF/e^(GG/HH*II)	12697	kg	JJ
Change in Mass	KK=FF-JJ	503	kg	KK

Step 6 Results: Using the TLI mass of the Falcon 9 Heavy of 13.2 metric tons and solving for Mass final results in an available payload mass of 12,697 kg.

Step 7: Calculate Launch Cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	LL
Number of Launches	LL=MM	4	Launches	MM
Cost per Launch	Figure of Merit	158	$M/Launch	NN
Launch Cost	OO=MM*NN	632	$M	OO

Step 7 Results: The total launch for 4 satellites is $632 million dollars. The Falcon 9 Heavy cost was used and even though the mass of the Falcon 9 Heavy is 13,200 kg which could allow the launch vehicle to carry 2 satellites, this thesis assumes that the volume of the SPS would use all of the Falcon 9 Heavy's volume.

Step 8: Calculate the total system cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	PP
Cost per Satellite	QQ=O	604	$M	QQ
Constellation Cost	RR=PP*QQ	2415	$M	RR
System Cost	SS=EE+OO+RR	3078	$M	SS

Step 8 Results: Using the cost of the lunar receiver, the launch cost, and the constellation cost, the total system cost for the laser solar power satellite concept is $3.1 billion dollars.

Table 12 shows the analysis when using the SLA Laser receiver, and as mentioned in the background information, the other receiver option was the VMJ receiver which has a higher conversion efficiency at 13.6 W/cm^2 (136,000 W/m^2) but a lower optical-to-electrical conversion efficiency of 23%. The result of using the VMJ receiver drastically impacts the total system cost due to the lower total end-to-end efficiency while decreasing the size of the lunar receiver from the exceptional conversion efficiency. This is not desirable given that the laser lunar receiver is not the cost driver to the laser system.

Table 12 shows the analysis of the Laser SPS concept when using the four-satellite frozen elliptical orbit, but as mentioned in the background information the L1 Lagrange point could be used for lunar South Pole coverage with a reduced constellation size to two satellites. This greatly reduces the total system cost. Using a L1 satellite to lunar receiver distance of 57,000 km, results in a change only to the satellite transmitter diameter. The resulting transmitter diameter when using the L1 Lagrange point for the laser SPS concept is a diameter of approximately 29 meters. The L1 Lagrange point is a viable option for reducing the total system cost and should be considered.

2. Sensitivity Analysis

The power thread analysis for the microwave and laser concepts is summarized in the Table 13 including key figures of merit. With the baseline figures of merit, the microwave per satellite cost is less than the laser per satellite cost given the increased end-to-end efficiency; however, the microwave total system cost is greater than the laser system cost because the microwave receiver is large and costly. Before giving a final conclusion on the feasibility and selection of an architecture, a sensitivity analysis on key parameters was conducted to show a comparison between the concepts.

Table 13. Microwave and Laser SPS Summary

	MICROWAVE	LASER	UNITS
FREQUENCY	94		GHz
WAVELENGTH		805	Nanometers (nm)
LUNAR OUTPOST POWER REQUIREMENT	100	100	kW
SATELLITE APOGEE	9881	9881	km
END-TO-END EFFICIENCY	11 %	8.3 %	%
SATELLITE POWER REQUIREMENT	912	1208	kW
SATELLITE POWER PER UNIT MASS	0.2	0.2	kW/kg
SATELLITE MASS	4.6	6.0	Metric Tons (mT)
SATELLITE COST PER UNIT MASS	100,000	100,000	$/kg
SATELLITE COST	456	604	$M
TRANSMITTER DIAMETER	277	5.45	Meters
RECEIVER DIAMETER	278	21	Meters
RECEIVER COST	970	30	$M
RECEIVER LUNAR COST PER UNIT MASS	100,000	100,000	$/kg
TOTAL LAUNCH VEHICLE COST	632	632	$M
LAUNCH VEHICLE MASS TO TLI	13.2	13.2	Metric Tons (mT)
CONSTELLATION SIZE	4	4	Satellites
TOTAL SYSTEM COST	3,425	3,078	$M

a. Microwave Receiver Sensitivity

Since the cost of the microwave receiver placed at the lunar outpost is driving the microwave system cost greater than the laser system, a sensitivity analysis was conducted on the receiver lunar cost per unit mass as seen in Table 14 and Figure 38. The receiver lunar cost per unit mass is the amount of money required to put a 1 kg on the Moon; this value was allowed to vary from the baseline value of $100,000 per kg to $50,000 per kg. From Figure 38, the microwave system becomes less expensive relative to the laser system when the receiver lunar cost per unit mass is below approximately $62,500 dollars per kg.

Table 14. Sensitivity of System Cost to a change in Lunar Receiver Cost per Unit Mass

		System Cost ($M)	
		Laser	Microwave
Lunar Receiver Cost per Unit Mass ($/kg)	$ 50,000	$ 3,063	$ 2,940
	$ 55,000	$ 3,064	$ 2,989
	$ 60,000	$ 3,066	$ 3,037
	$ 65,000	$ 3,067	$ 3,086
	$ 70,000	$ 3,069	$ 3,134
	$ 75,000	$ 3,070	$ 3,183
	$ 80,000	$ 3,072	$ 3,231
	$ 85,000	$ 3,073	$ 3,280
	$ 90,000	$ 3,075	$ 3,328
	$ 95,000	$ 3,076	$ 3,377
	$ 100,000	$ 3,078	$ 3,425

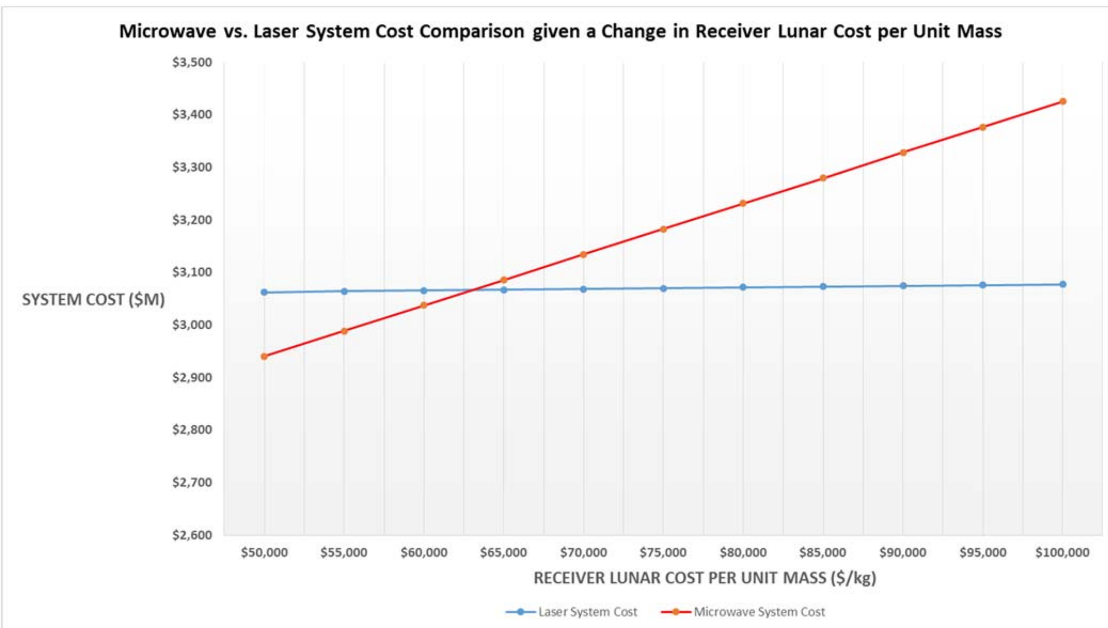

Figure 38. Microwave vs. Laser System Cost Comparison given a Change in Receiver Lunar Cost per Unit Mass

The microwave SPS system has a transmitter diameter of 277 meters to produce a receiver diameter of 278 meters. This is an exceptionally large satellite transmitter. If the satellite transmitter was at a maximum value of 22 meters as identified in the background research the resulting lunar receiver diameter size would be 3,500 meters. The resulting total system cost would be $156 billion dollars and the receiver lunar cost per unit mass would have to be below $400 per kg to be economically feasible against the laser system

(Appendix A). It is unlikely that the cost per kilogram to land on the Moon would ever be this low. Using the Microsoft Excel goal/seek function with the 22 meter transmitter diameter and with the lunar receiver cost at $600 million while allowing the spacecraft distance to vary, resulted in a spacecraft distance of only 617 kilometers and a total system cost of $3,055 million (Appendix B).

b. System Cost Sensitivity

A comparison of the system cost for the microwave and laser concepts was conducted while varying the power requirement versus SPS power per unit mass, and power requirement versus SPS cost per unit mass. As expected, when the SPS power per unit mass increases, the total system cost decreases as illustrated in Table 15 and Table 16. When the lunar outpost power requirement is increased 300 kW, the efficiency of both the microwave and laser concepts would have to increase their SPS power per unit mass to 0.35 kW/kg to remain under $5 billion for a total system cost. The rate of change given an increase in lunar outpost power requirement and SPS power per unit mass can be seen in Figure 39. When the power requirement at the lunar outpost is 100 kW, the laser concept remains the less expensive option across all SPS power per unit mass figures greater than 0.15 kW/kg. When the lunar base power requirement is 300 kW, the microwave system is the more economical option up to a SPS power per unit mass of 0.4 kW/kg due to the higher end-to-end efficiency.

Table 15. Microwave System Cost Sensitivity to a Change in Power Requirement and SPS Power per Unit Mass ($M)

Microwave System Cost Sensitivity to a change in Power Requirement and SPS Power per Unit Mass ($M)											
		Lunar Power Requirement (kW)									
		100	150	200	250	300	350	400	450	500	550
SPS Power Per Unit Mass (kW/kg)	0.1	5249	7072	8896	10719	12543	14366	16189	18013	19836	21660
	0.15	4033	5249	6464	7680	8896	10111	11327	12543	13758	14974
	0.2	3425	4337	5249	6160	7072	7984	8896	9807	10719	11631
	0.25	3060	3790	4519	5249	5978	6707	7437	8166	8896	9625
	0.3	2817	3425	4033	4641	5249	5856	6464	7072	7680	8288
	0.35	2644	3165	3686	4207	4728	5249	5770	6291	6812	7333
	0.4	2513	2969	3425	3881	4337	4793	5249	5704	6160	6616
	0.45	2412	2817	3223	3628	4033	4438	4843	5249	5654	6059
	0.5	2331	2696	3060	3425	3790	4155	4519	4884	5249	5613
	0.55	2265	2596	2928	3259	3591	3922	4254	4586	4917	5249

Legend	Range
Green	Less Than $5B
Yellow	Greater Than $5B but Less Than $8B
Red	Greater Than $8B

Table 16. Laser System Cost Sensitivity to a Change in Power Requirement and SPS Power per Unit Mass ($M)

Laser System Cost Sensitivity to a change in Power Requirement and SPS Power per Unit Mass ($M)											
		Lunar Power Requirement (kW)									
		100	150	200	250	300	350	400	450	500	550
SPS Power Per Unit Mass (kW/kg)	0.1	5493	7924	10354	12785	15215	17646	20076	22507	24937	27368
	0.15	3883	5508	7133	8759	10384	12010	13635	15260	16886	18511
	0.2	3078	4300	5523	6746	7969	9191	10414	11637	12860	14083
	0.25	2594	3576	4557	5538	6519	7501	8482	9463	10444	11426
	0.3	2272	3093	3913	4733	5553	6373	7194	8014	8834	9654
	0.35	2042	2748	3453	4158	4863	5568	6273	6979	7684	8389
	0.4	1870	2489	3108	3727	4346	4964	5583	6202	6821	7440
	0.45	1736	2287	2839	3391	3943	4495	5047	5598	6150	6702
	0.5	1628	2126	2625	3123	3621	4119	4617	5115	5613	6112
	0.55	1540	1995	2449	2903	3357	3812	4266	4720	5174	5628

Legend	Range
Green	Less Than $5B
Yellow	Greater Than $5B but Less Than $8B
Red	Greater Than $8B

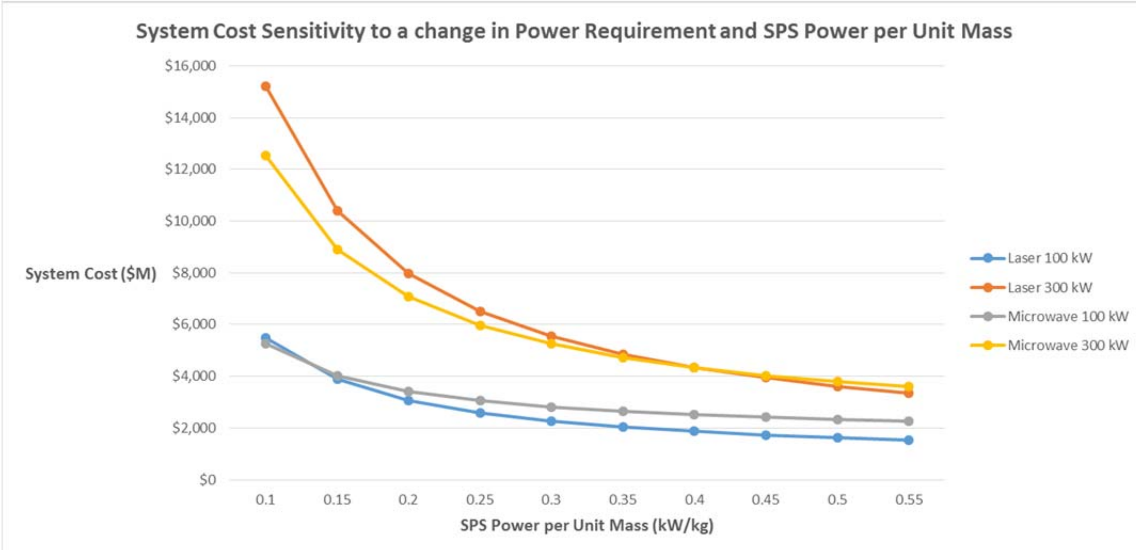

Figure 39. System Cost Sensitivity to a Change in Power Requirement and SPS Power per Unit Mass

A driving factor in the total system cost is the SPS cost per unit mass figure of merit which is the amount of money required to produce 1 kg of satellite mass. As mentioned in the background and as a point of reference, the AEHF satellite had a cost per unit mass of $156,000 per kg. Table 17 and Table 18 show sensitivity of the total system cost when the baseline SPS cost per unit mass is allowed to vary from $50,000 per kg to $140,000 per kg. If the lunar base power requirement was increased to 300 kW and the total system cost was expected to be under $5 billon, the SPS cost per unit mass for the microwave system would have to be less than $60,000 per kg and under $50,000 per kg for the laser system. From Figure 40, one can see that as the power requirement and SPS cost per unit mass increase, the microwave system is the less expensive option. This is driven by the microwave receiver cost being a relatively smaller percentage of the total cost and due to the higher microwave end-to-end efficiency.

Table 17. Microwave System Cost Sensitivity to a Change in Power Requirement and SPS Cost per Unit Mass ($M)

Microwave System Cost Sensitivity to a change in Power Requirement and SPS Cost per Unit Mass ($M)											
		Lunar Power Requirement (kW)									
		100	150	200	250	300	350	400	450	500	550
	$ 50,000	2513	2969	3425	3881	4337	4793	5249	5704	6160	6616
	$ 60,000	2696	3243	3790	4337	4884	5431	5978	6525	7072	7619
	$ 70,000	2878	3516	4155	4793	5431	6069	6707	7346	7984	8622
SPS Cost per	$ 80,000	3060	3790	4519	5249	5978	6707	7437	8166	8896	9625
Unit Mass	$ 90,000	3243	4063	4884	5704	6525	7346	8166	8987	9807	10628
($/kg)	$ 100,000	3425	4337	5249	6160	7072	7984	8896	9807	10719	11631
	$ 110,000	3607	4610	5613	6616	7619	8622	9625	10628	11631	12634
	$ 120,000	3790	4884	5978	7072	8166	9260	10354	11448	12543	13637
	$ 130,000	3972	5157	6343	7528	8713	9898	11084	12269	13454	14640
	$ 140,000	4155	5431	6707	7984	9260	10537	11813	13090	14366	15642

Legend	Range
Green	Less Than $5B
Yellow	Greater Than $5B but Less Than $8B
Red	Greater Than $8B

Table 18. Laser System Cost Sensitivity to a Change in Power Requirement and SPS Cost per Unit Mass ($M)

Laser System Cost Sensitivity to a change in Power Requirement and SPS Cost per Unit Mass ($M)											
		Lunar Power Requirement (kW)									
		100	150	200	250	300	350	400	450	500	550
	$ 50,000	1870	2489	3108	3727	4346	4964	5583	6202	6821	7440
	$ 60,000	2111	2851	3591	4330	5070	5810	6550	7289	8029	8769
	$ 70,000	2353	3213	4074	4934	5795	6655	7516	8376	9237	10097
SPS Cost per	$ 80,000	2594	3576	4557	5538	6519	7501	8482	9463	10444	11426
Unit Mass	$ 90,000	2836	3938	5040	6142	7244	8346	9448	10550	11652	12754
($/kg)	$ 100,000	3078	4300	5523	6746	7969	9191	10414	11637	12860	14083
	$ 110,000	3319	4663	6006	7350	8693	10037	11380	12724	14068	15411
	$ 120,000	3561	5025	6489	7954	9418	10882	12347	13811	15275	16740
	$ 130,000	3802	5387	6972	8558	10143	11728	13313	14898	16483	18068
	$ 140,000	4044	5750	7455	9161	10867	12573	14279	15985	17691	19397

Legend	Range
Green	Less Than $5B
Yellow	Greater Than $5B but Less Than $8B
Red	Greater Than $8B

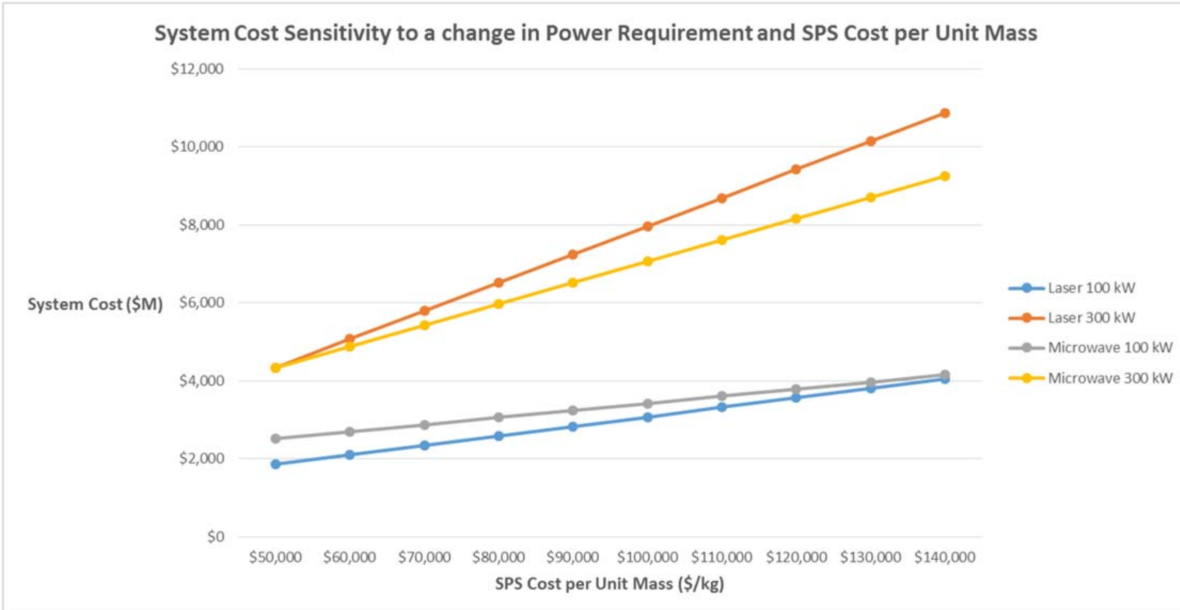

Figure 40. System Cost Sensitivity to a Change in Power Requirement and SPS Cost per Unit Mass

c. Laser SPS Sensitivity

Given the baseline figures of merit the laser system is the more economically feasible solution. From the system cost sensitivity, it is the better solution when the SPS power per unit mass increases and when the SPS cost per unit mass remain below $140,000 per kg for a 100 kW lunar power requirement. The ultimate conclusion is that the Laser SPS system even with a lower end-to-end efficiency is the better architecture. The Laser SPS system is the better architecture because the Microwave SPS total system cost is driven by the cost of the receiver placed on the Moon and the large microwave transmitter is an order of magnitude greater than the largest antenna placed in orbit. The microwave receiver size is determined by the apogee distance using the unique frozen elliptical orbit that is stable for a 10-year period which results in a receiver that is unacceptably large and costly. From this conclusion, a sensitivity analysis was conducted on the laser SPS system to determine the impact to the laser satellite mass as the parameters for lunar power requirement, SPS power per unit mass, and end-to-end efficiency varied.

Table 19 shows that the baseline solution for a SPS system with a 0.2 kW/kg power per unit mass and a lunar outpost power requirement of 100 kW produces a SPS mass of 6.0 metric tons (mT). Research into launch vehicle masses to a trans-lunar injection (TLI) orbit resulted in two launch vehicles, the Falcon 9 Heavy, and the SLS Block 1, which have a TLI maximum of 13.2 mT and 25 mT, respectively, and a TLI to LLO mass of 12.7 mT and 24 mT, respectively when using hall current thrusters. The laser satellite mass sensitivity showed that if lunar outpost power requirement is changed to 300 kW, this results in mass greater than the Falcon 9 Heavy capacity. The SLS Block 1 launch vehicle would have to be used if the power requirement increased to 250 kW and the SPS power per unit mass remained 0.2 kW/kg.

Table 19. Single Laser Satellite Mass Sensitivity to a Change in Power Requirement and SPS Power per Unit Mass (mT)

Single Satellite Mass Sensitivity to a change in Power Requirement and SPS Power per Unit Mass (mT)

		Lunar Power Requirement (kW)									
		100	**150**	**200**	**250**	**300**	**350**	**400**	**450**	**500**	**550**
SPS Power per Unit Mass (kW/kg)	**0.1**	12.1	18.1	24.2	30.2	36.2	42.3	48.3	54.3	60.4	66.4
	0.15	8.1	12.1	16.1	20.1	24.2	28.2	32.2	36.2	40.3	44.3
	0.2	6.0	9.1	12.1	15.1	18.1	21.1	24.2	27.2	30.2	33.2
	0.25	4.8	7.2	9.7	12.1	14.5	16.9	19.3	21.7	24.2	26.6
	0.3	4.0	6.0	8.1	10.1	12.1	14.1	16.1	18.1	20.1	22.1
	0.35	3.5	5.2	6.9	8.6	10.4	12.1	13.8	15.5	17.3	19.0
	0.4	3.0	4.5	6.0	7.5	9.1	10.6	12.1	13.6	15.1	16.6
	0.45	2.7	4.0	5.4	6.7	8.1	9.4	10.7	12.1	13.4	14.8
	0.5	2.4	3.6	4.8	6.0	7.2	8.5	9.7	10.9	12.1	13.3
	0.55	2.2	3.3	4.4	5.5	6.6	7.7	8.8	9.9	11.0	12.1

Legend	Range
Green	Less Than 12.7 mT
Yellow	Greater Than 12.7 mT but Less Than 24 mT
Red	Greater Than 24 mT

Table 20 shows that the baseline solution for a SPS system with a 0.2 kW/kg power per unit mass, and end to end efficiency of 10% and a lunar outpost power requirement of 100 kW produces a SPS mass of 5.0 metric tons (mT). However, with a 300 kW lunar outpost power requirement, the baseline SPS system solution exceeds the maximum Falcon 9 Heavy TLI to LLO mass of 12.7 mT when using hall current

thrusters. In order to use the Falcon 9 Heavy and produce 300 kW for the outpost, the end to end efficiency would have to be improved to 15%.

Table 20. Single Laser Satellite Mass Sensitivity to a Change in Power Requirement and End-to-End Efficiency (mT)

Single Satellite Mass Sensitivity to a change in Power Requirement and End-to-End Efficiency (mT)

		Lunar Power Requirement (kW)									
		100	**150**	**200**	**250**	**300**	**350**	**400**	**450**	**500**	**550**
End-to-End Efficiency (%)	**5%**	10.0	15.0	20.0	25.0	30.0	35.0	40.0	45.0	50.0	55.0
	10%	5.0	7.5	10.0	12.5	15.0	17.5	20.0	22.5	25.0	27.5
	15%	3.3	5.0	6.7	8.3	10.0	11.7	13.3	15.0	16.7	18.3
	20%	2.5	3.8	5.0	6.3	7.5	8.8	10.0	11.3	12.5	13.8
	25%	2.0	3.0	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0
	30%	1.7	2.5	3.3	4.2	5.0	5.8	6.7	7.5	8.3	9.2
	35%	1.4	2.1	2.9	3.6	4.3	5.0	5.7	6.4	7.1	7.9
	40%	1.3	1.9	2.5	3.1	3.8	4.4	5.0	5.6	6.3	6.9
	45%	1.1	1.7	2.2	2.8	3.3	3.9	4.4	5.0	5.6	6.1
	50%	1.0	1.5	2.0	2.5	3.0	3.5	4.0	4.5	5.0	5.5

Legend	**Range**
Green	Less Than 12.7 mT
Yellow	Greater Than 12.7 mT but Less Than 24 mT
Red	Greater Than 24 mT

In addition to the Laser SPS mass sensitivity, analysis was conducted using the Excel goal/seek function to determine how inefficient the laser SPS concept could be while still providing a 100 kW to the lunar outpost and use the Falcon 9 Heavy (Appendix C). The inefficiencies of the laser SPS could be from degradation to the satellite transmitter or photovoltaic panels or increased system mass from complexities in the thermal management system or in the pointing and tracking system. The goal/seek function set the SPS mass to 12.7 metric tons which is the available payload mass of the Falcon 9 Heavy using hall current thrusters while allowing the SPS power per unit mass to vary. The results of this sensitivity analysis showed that the SPS power per unit mass could be as low as 0.095 kW/kg and still use the Falcon 9 Heavy and provide a 100 kW to the lunar outpost. The resulting total system cost was $5.7 billion.

V. CONCLUSION AND RECOMMENDATIONS

A. SUMMARY OF WORK

This thesis began with the presumption that mankind is back on the journey to lunar exploration and will require a continuous source of power to enable that goal. The question this thesis tried to answer was this; Is it feasible to provide power to a lunar polar outpost using a satellite constellation in lunar orbit? To answer this question, it was necessary to conduct a literature review on several aspects of the problem to include the types of lunar orbits, the history and state of the technology for microwave and laser wireless power transmission, alternative ideas for lunar power generation, and the history and development of solar power satellite concepts. After identifying the lunar outpost site, power requirement, and constellation parameters, research required that a system cost, mass, and power thread calculation was conducted on two wireless power transmissions options using either microwave or laser. The end goal of this thesis was to setup a cost comparison to determine which wireless power transmission option was more feasible.

B. CONCLUSIONS

To effectively explore the Moon, astronauts will need a continuous source of power from either surface photovoltaics with complementary energy storage, nuclear power, or a solar power satellite. The analysis of alternatives showed the surface photovoltaics suffered from significant mass issues related to the energy storage system, and that the nuclear power option suffered from safety and political issues in the launch and landing risk. The photovoltaic solution using peaks of eternal light did reduce the mass issues and is a viable alternative to the SPS concept; however, this solution would need to be modeled for a feasibility comparison. It was determined from the system analysis that the laser SPS concept is feasible. The microwave SPS concept while technically feasible had significant challenges balancing the size and mass of the satellite transmitter and lunar receiver. The microwave SPS concept is both impractical when using the 277 meter transmitter diameter since this size is an order of magnitude greater

than the largest antenna put into orbit and economically impractical when using the 22 meter transmitter diameter as the receiver costs end up at $154 billion. Ultimately, the distance between the satellite transmitter and lunar receiver is too far for cost effective microwave wireless power transmission using the frozen elliptical orbit. The microwave sensitivity analysis when using Excel's goal/seek function showed that the viable spacecraft distance for microwave power transmission against the laser SPS concept is 617 kilometers. This distance would require dozens of satellites to provide full coverage to the lunar outpost.

The architecture tradespace included the location of the lunar outpost, method of power generation, type of lunar orbit, power requirement, constellation size, wireless power transmission option, and launch vehicle. For the architecture trade-offs, the South-Pole Aitkin Basin was chosen over the South Pole given the higher scientific value that could be accomplished for minimal increased system cost of one additional satellite. The power generation tradeoff was for a solar power satellite given the mass and safety constraints of the alternative. However, the solution of using peaks of eternal light is a viable alternative and would need to be modeled to completely understand the difference between the two solutions. The lunar orbit type tradeoff showed the use of a Halo orbit at the L1 LaGrange point was simply too far for microwave power transmission and resulted in extremely large satellite transmitter and lunar receiver sizes. The Halo orbit for the laser SPS concept resulted in a transmitter diameter of approximately 29 meters and should be considered a viable alternative to reduce the constellation size and therefore the total system cost. The tradespace for low lunar orbits allows for smaller microwave receiver sizes but increases the number of satellites needed to maintain coverage. The frozen elliptical orbit offered the greatest utility for lunar outpost coverage. The power requirement tradespace was related to the coverage provided by the constellation. A smaller constellation size could be used if the energy storage system at the lunar outpost provided power during small breaks in coverage. The analysis for the mass and cost of an energy storage system complementing a SPS constellation was not conducted and therefore the 100% coverage was required for analysis. The constellation size was determined using a 100% coverage requirement at the SPA Basin lunar outpost

which resulted in a minimal constellation size of four-satellites. The tradespace for the wireless power transmission between microwaves and lasers resulted in a higher per satellite cost for the laser system relative to the microwave satellite cost but led to a lower total system cost due the relatively expensive microwave receiver. Finally, the tradespace for the launch vehicle was conducted which was influenced by the mass of the satellite. Both wireless power transmission options with the lunar outpost power requirement of 100 kilowatts resulted in a small enough mass to ride on the Falcon 9 Heavy. The final architecture trades and the most feasible SPS solution can be seen in Figure 41. Selections are marked with green check marks.

Using total system cost to compare the feasibility of the SPS concepts resulted in a Microwave total system cost of $3,425 million dollars and a Laser total system cost of $3,078 million dollars. The Laser SPS concept is more feasible given this result.

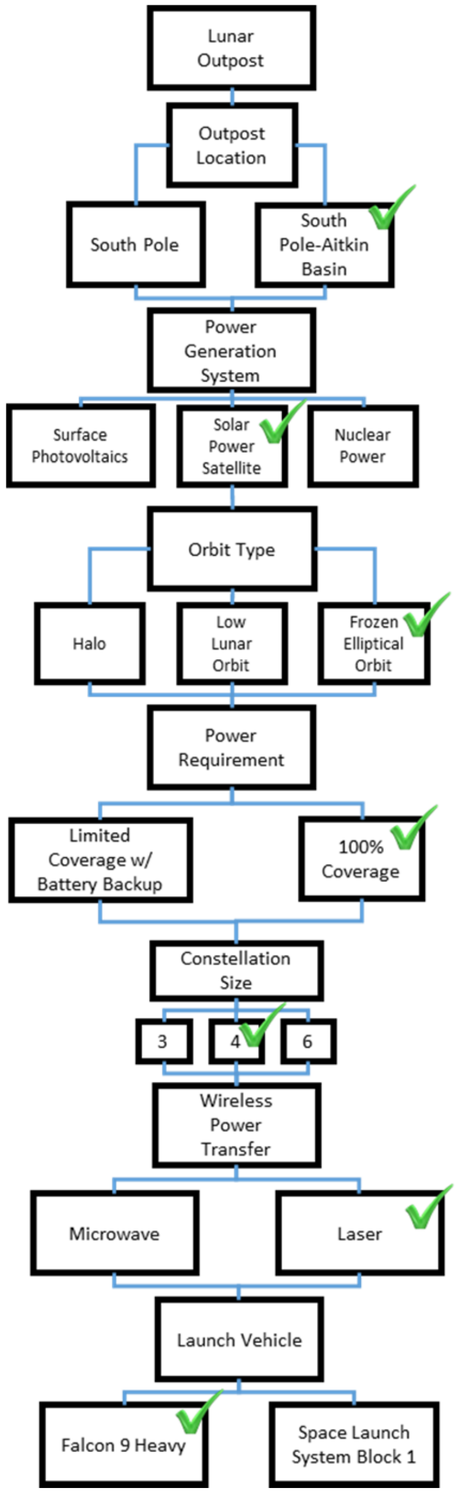

Figure 41. Selected Architecture Trades

C. AREAS FOR FUTURE WORK

There are several aspects of the SPS system providing power to a lunar orbit that need further development. An area of future study would be the analysis of a three-satellite constellation that uses a lunar gravitational field similar to the Earth's J2 affect to induce a controlled change in the right ascension of the ascending node (RAAN). This three-satellite constellation would be designed to have a RAAN drift of 2π radians per year. This would make the orbital plane rotate at the same speed as the Moon and achieve a fixed plane in the Sun-Earth-Moon system (Torres-Soto 2008). The goal would be to provide continuous coverage at the SPA Basin lunar outpost with a reduced constellation size of one less satellite. This analysis was not completed given the scope of the thesis and the limits of the moon gravitational model in STK.

Another area of future work includes the lunar dust environment. Recently, a dust cloud was discovered that is unevenly distributed around the Moon. The discovery was made from NASA's Lunar Atmosphere and Dust Environment Explorer (LADEE) which orbited the Moon for six months and documented over 140,000 dust impacts (Nicholes 2015). A lunar dust model would need to be created to understand the dust's impact on an optical system. The lunar dust could have an impact on the Laser SPS concept by introducing laser attenuation which will reduce the end-to-end efficiency and drive satellite mass and cost.

The most important area of future work is the increased fidelity to the SPS figures of merit. There was a significant amount of difficulty finding relevant figures of merit to model an entire SPS satellite. During research, a useful tool was discovered that was used on the Fresh Look Study, SERT study, and SPS-ALPHA study. That tool is called the Space Segment Model and the Integrated Architecture Assessment Model. These models were created for NASA by Science Applications International Corporation (SAIC) and the Futron Corporation with significant effort by Dr. Harvey Feingold (Feingold 2002). The models are more faithful to the architecture framework identified in the First International Assessment of Space Solar Power and provide increased fidelity to SPS calculations than the method presented here. These models should be released as open source code, and when new technology such as the SLA or VMJ receiver are developed,

their figures of merit could be included and compared against a baseline solution. Additional parametric modeling with more figures of merit is required to increase the fidelity of the solution. These additional figures of merit include the mean time between failure, satellite lifetime, and degradation to the lunar receiver, satellite transmitter, and photovoltaic panels.

The impact of batteries on the architecture is a necessary area of future work. As mentioned in the background and system architecture section the impact of an energy storage system was not directly calculated and compared against SPS concepts. For example, in the three-satellite constellation providing coverage to a SPA Basin lunar outpost, one can see in Figure 30 that it provides substantial coverage of approximately 90% with limited coverage of only a few days per month. Since the lunar outpost will require some amount of an energy storage capacity as a secondary backup to reduce mission risk, the three-satellite constellation could be applicable. The calculations for an energy storage system needs to be considered to reduce the total SPS system cost.

Finally, the last area of future work is the optimization of the architecture. The original idea for this thesis was to create an optimization problem in which given the architecture trades of outpost location, outpost location scientific value, orbit types, wireless power transfer options (microwave vs. laser), and launch vehicles, the goal would be to maximize the subjective value of scientific research while minimizing the cost. The global lunar landing site study identified many locations that provide scientific value which could be accessed by different and ideally lower altitude satellites. If cost was the driving factor, then an optimization model could identify which landing site locations offer the best return on investment.

APPENDIX A. MICROWAVE SYSTEM ANALYSIS WITH A 22 METER TRANSMITTER DIAMETER

Microwave System Analysis		
Constraits	Value	Units
Frequency	94	GHz
Wavelength	0.0032	m
Spacecraft Range	9881	km

Step 1: Calculate the end-to-end power efficiency and SPS power requirement

		Value	Units	
Lunar Outpost Power Requirement		100	kW	A
RF to DC Conversion Efficiency	Figure of Merit	72%	%	B
Power Required due to RF to DC Conversion Efficiency	C=A/B	139	kW	C
RF Collection Efficiency	Figure of Merit	93%	%	D
Power Required due to RF Collection Collection Efficiency	E=C/D	149	kW	E
Amount of Power in the Main Lobe of the Beam	Figure of Merit	84%	%	F
Power Required due to Power in the Main Lobe of the Beam	G=E/F	178	kW	G
DC to RF Conversion Efficiency	Figure of Merit	78%	%	H
Power Required due to DC to RF Conversion Efficiency	I=G/H	228	kW	I
Solar to DC Conversion Efficiency	Figure of Merit	25%	%	J
Power Required due to Solar to DC Conversion Efficiency	K=I/J	912	kW	K
End-to-End Power Efficiency	L=B*D*F*H*J	11.0%	%	L
SPS Power Requirement	M=K	912	kW	M

Step 1 Results: The end-to-end efficiency for the microwave power thread is 11% which results in a SPS power requirement of 912 kilowatts to deliver 100 kilowatts to the lunar outpost.

Step 2: Calculate the mass and cost of the SPS.

		Value	Units	
SPS Power per Unit Mass	Figure of Merit	0.2	kW/kg	N
SPS Mass	O=M/N	4.6	mT	O
SPS Cost per Unit Mass	Figure of Merit	100,000	$/kg	P
SPS Satellite Cost	Q=O*P	456	$M	Q

Step 3: Calculate the optimum size of the microwave receiver

	Value	Units	
Conversion Density	1	Watt/cm^2	R
1 m^2	10,000	cm^2	S
E in Watts	149343	Watts	T
U=T/(R*S)	15	m^2	U

Step 3 Result: The optimum microwave receiver area is 15 m^2

Step 4: Calculate the size of the SPS transmitter given the optimum size of the microwave receiver					
			Value	Units	
	Receiver Diameter	V=SQRT(U*4/π)	4.4	m	V
		Spacecraft Apogee	9881	km	W
		Frequency	94	GHz	X
		Speed of Light	3.00E+08	m/s	Y
	Wavelength	Z=Y/X	0.0032	m	Z
Using Main Lobe Diffraction Equation	Transmitter Diameter	2.44=(V*AA)/(Z*W)	17646	m	AA

Step 4 Results: Using the optimum size of the lunar receiver results in an unacceptable large transmitter diamter of 17,646 meters. The largest antenna launched on the SkyTerra-1 satellite will be used of 22 meters.

Step 4.1: Using the largest antenna of 22 meters launched on the SkyTerra-1 satellite					
			Value	Units	
	Transmitter Diameter		22	m	DD
Using Main Lobe Diffraction Equation	Receiver Diamter	2.44=(P*R)/(N*K)	3498	m	EE

Step 4.2 Results: The 22 meter transmitter diameter results in a lunar receiver diameter of 3500 meters.

Step 5: Calculate the mass and cost of the lunar receiver using the 278 meter diameter				
		Value	Units	
Receiver Area	FF=π*(CC/2)^2	9607553	m^2	FF
Receiver Mass per Unit Area	Figure of Merit	0.16	kg/m^2	GG
Receiver Mass	HH=FF*GG	1537208	kg	HH
Receiver Lunar Cost per Unit Mass	Figure of Merit	100,000	$/kg	II
Receiver Lunar Cost	JJ=HH*II	153721	$M	JJ

Step 5 Results: Given the balance between the size of the SPS transmitter and lunar receiver, the cost of the receiver is $154 billion.

Step 6: Calculate Trans-Lunar Orbit (TLI) to Low Lunar Orbit (LLO)				
		Value	Units	
Mass Initial	Figure of Merit	13200	kg	KK
Delta-V	Figure of Merit	800	m/s	LL
Specific Impulse	Figure of Merit	2100	s	MM
Earth Gravitational Constant	Constant	9.807	m/s^2	NN
Mass Final	OO=KK/e^(LL/MM*NN)	12697	kg	OO
Change in Mass	PP=KK-OO	503	kg	PP

Results: Using the TLI mass of the Falcon 9 Heavy of 13.2 metric tons and solving for Mass final results in an available payload mass of 12,697 kg.

Step 7: Calculate Launch Cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	QQ
Number of Launches	RR=QQ	4	Launches	RR
Cost per Launch	Figure of Merit	158	$M/Launch	SS
Launch Cost	TT=RR*SS	632	$M	TT

Step 6 Results: The total launch cost for 4 satellites is $632 million dollars. The Falcon 9 Heavy cost was used and even though the mass of the Falcon 9 Heavy is 13,200 kg which could allow the launch vehicle to carry 2 satellites, this thesis assumes that the volume of the SPS especially given a 277 meter diameter transmitter, would use all of the Falcon 9 Heavy's volume.

Step 8: Calculate the total system cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	UU
Cost per Satellite	VV=Q	456	$M	VV
Constellation Cost	WW=UU*VV	1823	$M	WW
System Cost	XX=JJ+TT+QQ	156176	$M	XX

Step 7 Results: Using the cost of the lunar receiver, the launch cost, and the constellation cost, the total system cost for the microwave solar power satellite concept is $156 billion dollars.

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APPENDIX B. MICROWAVE SYSTEM ANALYSIS USING GOAL SEEK TO LIMIT SPACECRAFT DISTANCE

Microwave System Analysis

Constraits	Value	Units
Frequency	94	GHz
Wavelength	0.0032	m
Spacecraft Range	617	km

Step 1: Calculate the end-to-end power efficiency and SPS power requirement

		Value	Units	
Lunar Outpost Power Requirement		100	kW	A
RF to DC Conversion Efficiency	Figure of Merit	72%	%	B
Power Required due to RF to DC Conversion Efficiency	C=A/B	139	kW	C
RF Collection Efficiency	Figure of Merit	93%	%	D
Power Required due to RF Collection Collection Efficiency	E=C/D	149	kW	E
Amount of Power in the Main Lobe of the Beam	Figure of Merit	84%	%	F
Power Required due to Power in the Main Lobe of the Beam	G=E/F	178	kW	G
DC to RF Conversion Efficiency	Figure of Merit	78%	%	H
Power Required due to DC to RF Conversion Efficiency	I=G/H	228	kW	I
Solar to DC Conversion Efficiency	Figure of Merit	25%	%	J
Power Required due to Solar to DC Conversion Efficiency	K=I/J	912	kW	K
End-to-End Power Efficiency	L=B*D*F*H*J	11.0%	%	L
SPS Power Requirement	M=K	912	kW	M

Step 1 Results: The end-to-end efficiency for the microwave power thread is 11% which results in a SPS power requirement of 912 kilowatts to deliver 100 kilowatts to the lunar outpost.

Step 2: Calculate the mass and cost of the SPS.

		Value	Units	
SPS Power per Unit Mass	Figure of Merit	0.2	kW/kg	N
SPS Mass	O=M/N	4.6	mT	O
SPS Cost per Unit Mass	Figure of Merit	100,000	$/kg	P
SPS Satellite Cost	Q=O*P	456	$M	Q

Step 3: Calculate the optimum size of the microwave receiver			Value	Units	
		Conversion Density	1	Watt/cm^2	R
		1 m^2	10,000	cm^2	S
		E in Watts	149343	Watts	T
		U=T/(R*S)	15	m^2	U

Step 3 Result: The optimum microwave receiver area is 15 m^2

Step 4: Calculate the size of the SPS transmitter given the optimum size of the microwave receiver			Value	Units	
	Receiver Diameter	V=SQRT(U*4/π)	4.4	m	V
		Spacecraft Apogee	617	km	W
		Frequency	94	GHz	X
		Speed of Light	3.00E+08	m/s	Y
	Wavelength	Z=Y/X	0.0032	m	Z
Using Main Lobe Diffraction Equation	Transmitter Diameter	2.44=(V*AA)/(Z*W)	1102	m	AA

Step 4 Results: Using the optimum size of the lunar receiver results in an unacceptable large transmitter diamter of 17,646 meters. The largest antenna launched on the SkyTerra-1 satellite will be used of 22 meters.

Step 4.1: Using the largest antenna of 22 meters launched on the SkyTerra-1 satellite			Value	Units	
	Transmitter Diameter		22	m	DD
Using Main Lobe Diffraction Equation	Receiver Diamter	2.44=(P*R)/(N*K)	219	m	EE

Step 4.2 Results: The 22 meter transmitter diameter results in a lunar receiver diameter of 3500 meters.

Step 5: Calculate the mass and cost of the lunar receiver using the 278 meter diameter		Value	Units	
Receiver Area	FF=π*(CC/2)^2	37500	m^2	FF
Receiver Mass per Unit Area	Figure of Merit	0.16	kg/m^2	GG
Receiver Mass	HH=FF*GG	6000	kg	HH
Receiver Lunar Cost per Unit Mass	Figure of Merit	100,000	$/kg	II
Receiver Lunar Cost	JJ=HH*II	600	$M	JJ

Step 5 Results: Given the balance between the size of the SPS transmitter and lunar receiver, the cost of the receiver is $970 million which is more than the cost for a single SPS satellite

Step 6: Calculate Trans-Lunar Orbit (TLI) to Low Lunar Orbit (LLO)

		Value	Units	
Mass Initial	Figure of Merit	13200	kg	KK
Delta-V	Figure of Merit	800	m/s	LL
Specific Impulse	Figure of Merit	2100	s	MM
Earth Gravitational Constant	Constant	9.807	m/s^2	NN
Mass Final	OO=KK/e^(LL/MM*NN)	12697	kg	OO
Change in Mass	PP=KK-OO	503	kg	PP

Results: Using the TLI mass of the Falcon 9 Heavy of 13.2 metric tons and solving for Mass final results in an available payload mass of 12,697 kg.

Step 7: Calculate Launch Cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	QQ
Number of Launches	RR=QQ	4	Launches	RR
Cost per Launch	Figure of Merit	158	$M/Launch	SS
Launch Cost	TT=RR*SS	632	$M	TT

Step 6 Results: The total launch cost for 4 satellites is $632 million dollars. The Falcon 9 Heavy cost was used and even though the mass of the Falcon 9 Heavy is 13,200 kg which could allow the launch vehicle to carry 2 satellites, this thesis assumes that the volume of the SPS especially given a 277 meter diameter transmitter, would use all of the Falcon 9 Heavy's volume.

Step 8: Calculate the total system cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	UU
Cost per Satellite	VV=Q	456	$M	VV
Constellation Cost	WW=UU*VV	1823	$M	WW
System Cost	XX=JJ+TT+QQ	3055	$M	XX

Step 7 Results: Using the cost of the lunar receiver, the launch cost, and the constellation cost, the total system cost for the microwave solar power satellite concept is $156 billion dollars.

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APPENDIX C. LASER SYSTEM ANALYSIS USING GOAL SEEK

Laser System Analysis

Constraits	Value	Units
Wavelength	805	m
Spacecraft Range	9881	km

Step 1: Calculate the end-to-end power efficiency and SPS power requirement

		Value	Units	
Lunar Outpost Power Requirement		100	kW	A
Laser to DC Conversion Efficiency	Figure of Merit	45%	%	B
Power Required due to Laser to DC Conversion Efficiency	C=A/B	222	kW	C
Laser Collection Efficiency	Figure of Merit	92%	%	D
Power Required due to Laser Collection Efficiency	E=C/D	242	kW	E
DC to Laser Conversion Efficiency	Figure of Merit	80%	%	F
Power Required due to DC to Laser Conversion Efficiency	G=E/F	302	kW	G
Solar to DC Conversion Efficiency	Figure of Merit	25%	%	H
Power Required due to Solar to DC Conversion Efficiency	I=G/H	1208	kW	I
End-to-End Power Efficiency	J=B*D*F*H	8.3%	%	J
SPS Power Requirement	K=I	1208	kW	K

Step 1 Results: The end-to-end efficiency for the laser power thread is 8.3% which results in a SPS power requirement of 1208 kilowatts to deliver 100 kilowatts to the lunar outpost.

Step 2: Calculate the mass and cost of the SPS.

		Value	Units	
SPS Power per Unit Mass	Figure of Merit	0.0951	kW/kg	L
SPS Mass	M=K/L	12.697	mT	M
SPS Cost per Unit Mass	Figure of Merit	100,000	$/kg	N
SPS Satellite Cost	O=M*N	1270	$M	O

Step 3: Calculate the size of the laser receiver

	Value	Units	
Conversion Density	0.069	Watt/cm^2	P
1 m^2	10,000	cm^2	Q
E in Watts	241546	Watts	R
S=R/(P*Q)	350	m^2	S

Step 3 Result: The laser receiver is 350 m^2

Step 4: Calculate the size of the SPS transmitter radius given the size of the laser receiver

			Value	Units	
	Beam Waist Radius		1.0	m	V
	Wavelength		805	nm	W
	Spacecraft Apogee		9881	km	X
Transmitter Radius	Y=V*SQRT(1+(W*X/(PI()*V^2))^2)		2.72	m	Y
	Transmitter Area	Z=PI()*r^2	23	m^2	Z

Step 4 Results: Given the laser receiver area and the radius of the laser beam waist, the transmitter radius is 2.72 meters with a total area of 23 m^2

Step 5: Calculate the mass and cost of the lunar receiver

		Value	Units	
Receiver Area	AA=S	350	m^2	AA
Receiver Mass per Unit Area	Figure of Merit	0.86	kg/m^2	BB
Receiver Mass	CC=AA*BB	301	kg	CC
Receiver Lunar Cost per Unit Mass	Figure of Merit	100,000	$/kg	DD
Receiver Lunar Cost	EE=CC*DD	30	$M	EE

Step 5 Results: Both the lunar receive and transmitter diameters are reasonable which leads to a receiver cost of $30 million dollars

Step 6: Calculate Trans-Lunar Orbit (TLI) to Low Lunar Orbit (LLO)

		Value	Units	
Mass Initial	Figure of Merit	13200	kg	FF
Delta-V	Figure of Merit	800	m/s	GG
Specific Impulse	Figure of Merit	2100	s	HH
Earth Gravitational Constant	Constant	9.807	m/s^2	II
Mass Final	JJ=FF/e^(GG/HH*II)	12697	kg	JJ
Change in Mass	KK=FF-JJ	503	kg	KK

Step 6 Results: Using the TLI mass of the Falcon 9 Heavy of 13.2 metric tons and solving for Mass final results in an available payload mass of 12,697 kg.

Step 7: Calculate Launch Cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	LL
Number of Launches	LL=MM	4	Launches	MM
Cost per Launch	Figure of Merit	158	$M/Launch	NN
Launch Cost	OO=MM*NN	632	$M	OO

Step 7 Results: The total launch for 4 satellites is $632 million dollars. The Falcon 9 Heavy cost was used and even though the mass of the Falcon 9 Heavy is 13,200 kg which could allow the launch vehicle to carry 2 satellites, this thesis assumes that the volume of the SPS would use all of the Falcon 9 Heavy's volume.

Step 8: Calculate the total system cost

		Value	Units	
Number of Satellites	Orbital Analysis	4	Satellites	PP
Cost per Satellite	QQ=O	1270	$M	QQ
Constellation Cost	RR=PP*QQ	5079	$M	RR
System Cost	SS=EE+OO+RR	5741	$M	SS

Step 8 Results: Using the cost of the lunar receiver, the launch cost, and the constellation cost, the total system cost for the laser solar power satellite concept is $5.2 billion dollars.

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